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FINAL REPORT – PHASE II

PROPELLANT SELECTION FOR UNMANNED SPACECRAFT PROPULSION SYSTEMS

CONTRACT NASw-1644

VOLUME I
RESULTS, CONCLUSIONS, AND RECOMMENDATIONS

PREPARED FOR
NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
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WASHINGTON, D.C.

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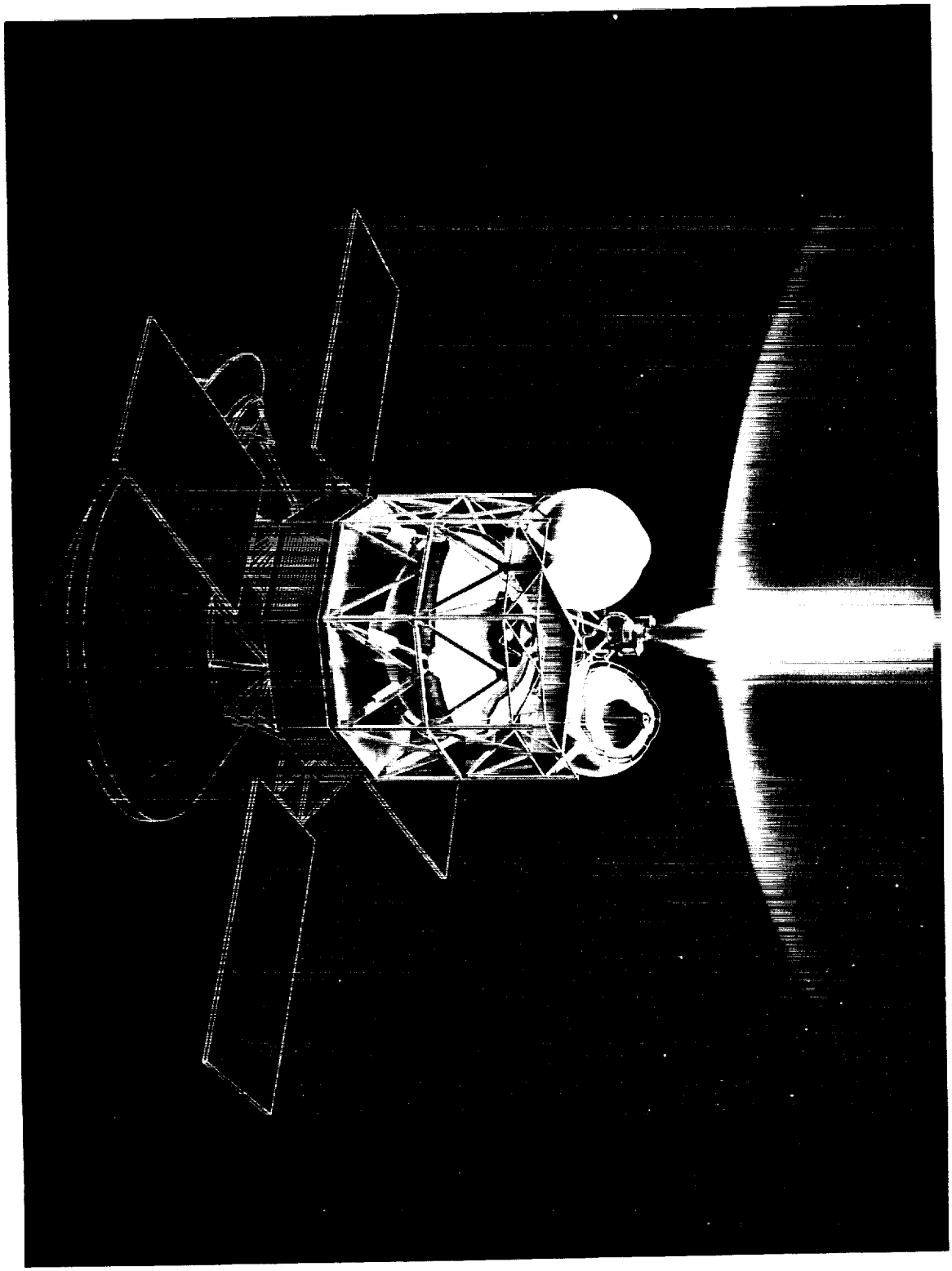

Commonality Stage – F_2/H_2 Concept

FOREWORD

This report was prepared by the Lockheed Missiles & Space Company, Sunnyvale, California, and contains the results of a study performed for the National Aeronautics and Space Administration, Office of Advanced Research and Technology, under Phase II of Contract NASw-1644, Propellant Selection for Unmanned Spacecraft Propulsion Systems. The report is printed in three volumes:

Volume I	Results, Conclusions, and Recommendations
Volume II	Analysis of Propellant Sensitivity, Secondary Propulsion, and Ground Operations
Volume III	Study of Propulsion Stage Commonality and Attitude Control Systems Requirements

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Section 1
SUMMARY

A detailed comparison of the storability and performance of high-energy liquid propellants was performed for a wide range of near-earth and interplanetary missions. Emphasis was placed on detailed thermodynamics, pressurization, and structural analyses of representative vehicle systems, and identification of related technology development requirements.

In Phase I, preliminary analysis was conducted on a broad range of missions, and was followed by a refined analysis of two systems. The two systems investigated in depth were the propulsion stage for an unmanned Mars Orbiter and the ascent stage for a Manned Mars Excursion Module (MEM) with fixed payloads. Specific propellants investigated were F_2/H_2, O_2/H_2, FLOX/CH_4, OF_2/CH_4, F_2/NH_3, OF_2/B_2H_6, $C\ell F_5$/MHF-5, and N_2O_4/A-50.

In Phase II the study of propellant sensitivity to thermodynamic and system design variables was carried to still greater depths, with particular emphasis on the sensitivity of propellant choice for a common propulsion stage performing alternate missions.

Major conclusions of the study are as follows:

- All candidate propellants can be stored nonvented for a wide range of probe, orbiter, and lander missions
- Propellant specific impulse is the predominant factor in propellant/system performance
- Fluorine/hydrogen provides the best performance for all mission applications studied
- Space-storable propellants provide high performance and are thermally less sensitive than hydrogen

- Spacecraft orientation constraints can be accepted since shadow shields are very effective
- Space-storable propellant feed systems simplicity is similar to earth-storables
- The concept of a commonality stage is very attractive for either F_2/H_2 or space-storable propellants
- All propellants studied can be loaded off-pad and maintained non-vented by active thermal control until launch

Technology development requirements compiled during the study include:

- Develop efficient, condensation-free ground thermal conditioning systems
- Demonstrate propellant tank and component materials compatibility with alternate propellants
- Demonstrate flexible, low weight, economical, low absorptivity/emissivity surface coatings
- Develop evacuated insulation systems for use in planetary atmospheres
- Develop leak-free electrical feed-throughs for fluorinated oxidizers
- Demonstrate leak-free fluorinated oxidizer storage and pressurization systems
- Develop sensors for fluorinated oxidizer management
- Develop low-flow, high-ΔP fuel pumps and small turbines
- Continue system tests to establish attainable performance of alternate propellants
- Develop optimum engine cooling techniques for each propellant
- Demonstrate deep throttling, restart, and control of small engines
- Develop devices for sensing propellant partial pressure

Section 2
INTRODUCTION

This report presents the results of work performed by the Lockheed Missiles & Space Company for Headquarters, NASA-OART, under Phase II of Contract NASw-1644 during the period Sept. 5, 1968 to June 6, 1969. The contract was directed by a Management Committee representing NASA Headquarters, Lewis Research Center, Marshall Space Flight Center, Manned Space Center, and the Jet Propulsion Laboratory.

For the purposes of this study, three general classes of liquid propellants were defined as follows:

- Earth storable: In the liquid state at earth ambient temperatures and pressures
- Cryogenic or deep cryogenic: Propellants using liquid hydrogen as the fuel
- Space storable: In the liquid state at temperatures below earth ambient but higher than liquid hydrogen

The Phase I study, documented in Ref. 1, was directed toward evaluating the performance of space-storable propellants in comparison with earth-storables and cryogens and emphasized identifying technology requirements related to thermal storage of propellants having attractive performance.

The Phase II study, documented in Ref. 2 which includes this volume, was designed to:

1. Provide greater depth of understanding of propellant sensitivity to thermodynamic and system design variables
2. Compare alternate approaches to accomplishing secondary propulsion maneuvers
3. Analyze the sensitivity of propellant choice for a common stage performing alternate missions

4. Analyze ground operations and facilities requirements as a function of propellant
5. Define attitude control system thruster requirements for alternate spacecraft and missions

Missions and stages analyzed were selected as items of convenience, and were not intended to establish configurations for any specific NASA program requirement.

2.1 STUDY APPROACH

The propellant study was performed in two major phases. Phase I was divided into three parts: (1) a mission investigation was performed to identify potential applications for space-storable propellants and to identify attractive candidates for a detailed stage investigation; (2) a detailed thermal, structural, and operational analysis was performed on two propulsion stages selected by NASA. Alternate earth-storable, space-storable, and cryogenic propellants were analyzed with emphasis on investigation of propellant storage requirements; (3) the results of (1) and (2) were used in developing propellant selection factors, in identifying areas of propulsion systems commonality, and in identifying major problem areas and technology development requirements resulting from the proposed use of space-storable propellants. Basic analysis assumptions for Phase I are shown in Table 1.

Phase II, designed to study new areas of interest and to answer questions raised during Phase I, was performed as a series of independent, or loosely interrelated, tasks and subtasks. These tasks included: (1) additional study of propellant selection sensitivity; (2) study of alternate modes for secondary propulsion; (3) study of the influence of propellant choice on the performance of a common stage, sized for a Mars Orbiter to be launched by Titan IIID/Centaur, and flown on alternate unmanned space missions; (4) study of ground operations and facilities problems and requirements associated with the alternate propellants, common stage, and Titan launch complex; (5) study of attitude control system requirements for spacecraft matched to the common stage.

A summary of the study elements, for both phases, is presented in Fig. 1.

Table 1

BASIC ANALYSIS ASSUMPTIONS FOR PHASE I

Parameter	Preliminary Analysis Assumptions	Refined Analysis Assumptions
Missions	Mission Spectrum	Mars Orbiter and MEM
Stage Inert Weight	Scaling Law Weights	Calculated Weights
Engine Weight	Scaling Law Weights	Engine Company Data
Tank Heat Input	Converted to Boiloff	Considered in Optimization
Stage Optimization	Boiloff and Insulation Only	Full System Optimization
Superinsulation Conductivity	$H_2 - 2.5 \times 10^{-5}$ Btu/hr-ft-°R O_2, FLOX, OF_2, F_2, $CH_4 - 5.0 \times 10^{-5}$ Btu/hr-ft-°R $NH_3 - 10.0 \times 10^{-5}$	
Surface Coatings (Nominal)	H_2, O_2, F_2, FLOX, CH_4, OF_2 – Optical Solar Reflector, α_s/ϵ = 0.05/0.80 NH_3 – Degraded White Thermatrol, α_s/ϵ = 0.30/0.95 N_2O_4/A-50, ClF_5, MHF-5 – Degraded White Skyspar, α_s/ϵ = 0.60/0.91	

2.2 STUDY GROUND RULES AND CONSTRAINTS

The following ground rules were used in the study:

- To achieve a common technical basis, as well as to expedite the analysis, the following scaling relationships, environmental models, and parametric mission models were used in Phase I, Part 1, Mission Analysis:
  - Scaling laws for propellant tank inert weights and any desirable propellant density
  - Thermal and micrometeorite environmental models
  - Thermal insulation and boiloff penalties

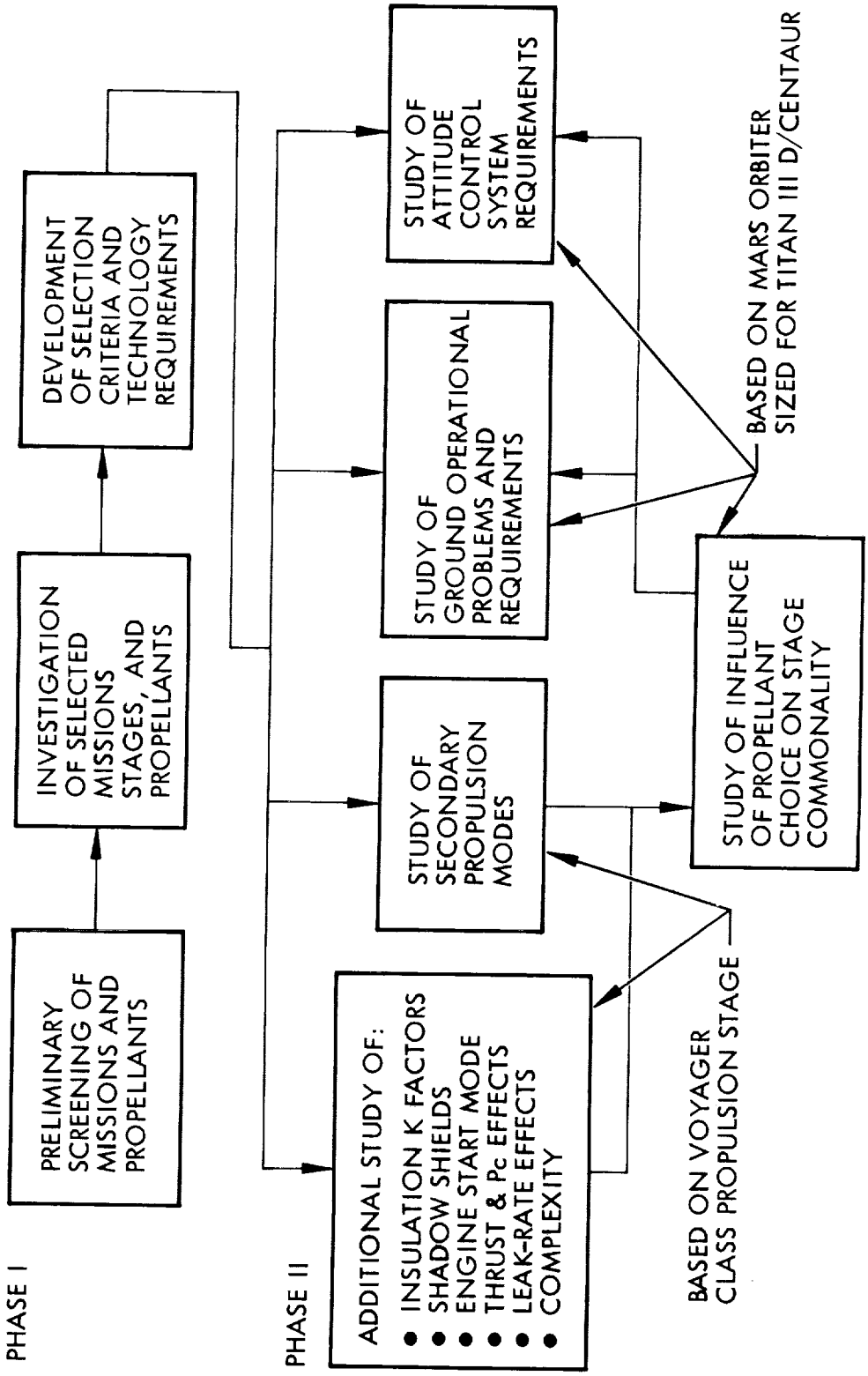
PHASE I
PRELIMINARY SCREENING OF MISSIONS AND PROPELLANTS
INVESTIGATION OF SELECTED MISSIONS STAGES, AND PROPELLANTS
DEVELOPMENT OF SELECTION CRITERIA AND TECHNOLOGY REQUIREMENTS
PHASE II
ADDITIONAL STUDY OF:
• INSULATION K FACTORS
• SHADOW SHIELDS
• ENGINE START MODE
• THRUST & Pc EFFECTS
• LEAK-RATE EFFECTS
• COMPLEXITY
BASED ON VOYAGER CLASS PROPULSION STAGE
STUDY OF SECONDARY PROPULSION MODES
STUDY OF GROUND OPERATIONAL PROBLEMS AND REQUIREMENTS
STUDY OF ATTITUDE CONTROL SYSTEM REQUIREMENTS
STUDY OF INFLUENCE OF PROPELLANT CHOICE ON STAGE COMMONALITY
BASED ON MARS ORBITER SIZED FOR TITAN III D/CENTAUR

Fig. 1 Study Elements

- Meteoroid shielding penalties
- Characteristic velocity requirements as a function of trip times

- The propellants used in Phase I, Part 1 were representative of the cryogenics, space storables, and earth storables. Specifically, they were F_2/H_2, O_2/H_2, $FLOX/CH_4$, F_2/NH_3, and N_2O_4/A-50.
- Evaluation procedures used in all analyses following Phase I, Part 1 included realistic calculations of heat leak through insulation, radiation interchange between surfaces, and conduction through insulation penetrations, down supports, and through the structure.

The following study constraints were specified by NASA:

- The space-storable propellants to be considered to the extent necessary following Phase I, Part 1 include:

  OF_2/CH_4
  $FLOX/CH_4$ (82.5 Percent F_2/17.5 Percent O_2)
  OF_2/B_2H_6 (Mars Orbiter Only)
  F_2/NH_3
  O_2/H_2 and Subcooled O_2/H_2
  F_2/H_2
  N_2O_4/A-50 (Mars Orbiter Only)
  ClF_5/MHF-5

  Selections within this list will be made for each subtask.
- Only pressure-fed engine systems are considered for OF_2/B_2H_6. Both pressure-fed and pump-fed engine systems (regeneratively cooled, ablatively cooled, or transpiration cooled) may be considered for the other fuels.
- All propulsion systems are assumed to have equal reliability.
- Sterilization criteria are not considered.
- Mission parameters and nonpropulsive portions of spacecraft vehicles are not studied in detail; these items are adopted from an original study for the various missions.
- Use of refrigeration systems is not considered.

2.3 LIMITATIONS OF ANALYSIS

The following limitations of the analyses are presented as an aid in understanding the study results:

- The study was propulsion-technology oriented.
- The analyses were heavily weighted toward propellant storage requirements, detailed thermal/structural/pressurization optimization, and performance comparisons.
- All designs featured separate fuel and oxidizer tanks. No analyses were made of tanks with common bulkheads.
- Oxidizer and fuel tanks were optimized separately as to pressure, tank weight, and insulation weight. This implies a need for mixture ratio control for pressure-fed systems.
- Subsystems, except for solar panels on the Mars Orbiter, were assumed separated from the propulsion module by a thermal barrier. Designs requiring integration of spacecraft equipment and propellant tankage would pose a more severe thermal storage problem for the space-storable and cryogenic propellants.
- Detailed analysis of propellant tank exposure was limited to extremes of exposure to the sun and complete shielding behind the Mars Orbiter capsule. Local shadow shielding was examined in lesser detail.
- No shadow shield studies were made for the MEM on the surface of Mars. Stage I tanks were assumed exposed to the sun.
- No general agreement could be found among rocket engine companies as to problems or merits of one propellant combination over another when used in an engine system, or as to the desirability and potential reliability of (1) pump-fed systems as compared to pressure-fed systems, and of (2) regenerative cooling versus transpiration cooling versus film and ablative cooling.
- No analysis was made of meteoroid shields discarded prior to stage firing since the stages analyzed required multiple firings.
- Analysis of differential boiloff of fluorine from FLOX was limited to ullage volume since fluorinated oxidizers were not vented.

Section 3
RESULTS – PHASE I

3.1 PRELIMINARY SCREENING OF MISSIONS AND PROPELLANTS

Early in Phase I, a systematic analysis was made in which a mssion matrix was established, a preliminary screening was conducted, and the selected missions were then evaluated through a preliminary thermodynamic and propulsion system analysis. The evaluation procedure is illustrated in Fig. 2. Scaling laws were used to conduct the performance analysis, which was accomplished using a computer program entitled Rapid Analysis of Propellants for Initial Design (RAPID).

The mission analysis was conducted for a broad spectrum of space missions. All of the propulsive steps in each mission were considered, and those having a $\Delta V \geq 3,500$ ft/sec were selected for analysis since they provided the possibility of increasing the stage performance by 10 percent or more when space-storable propellants were substituted for the earth storable N_2O_4/A-50. The missions and stages selected for analysis are listed in Table 2, together with important mission parameters.

Representative cryogenic, space-storable, and earth-storable propellants were selected for the initial mission screening analysis. The propulsion assumptions used for Phase I, Part 1 are shown in Table 3. These data were based on information available at the start of the study and were modified for Phase I, Part 2 as described in subsequent paragraphs. In Table 4, the propulsion stage initial weights for the various missions are normalized using N_2O_4/A-50 as the reference propellant. From Table 4 it is evident that, based on the scaling law analysis, a FLOX/CH_4 or F_2/NH_3 stage is from 20 to 45 percent lighter than the earth-storable stage to perform a specified mission. An O_2/H_2 stage is slightly heavier than FLOX/CH_4 for all but the large planet-departure stages, while F_2/H_2 results in the lightest stage for all but the long-duration

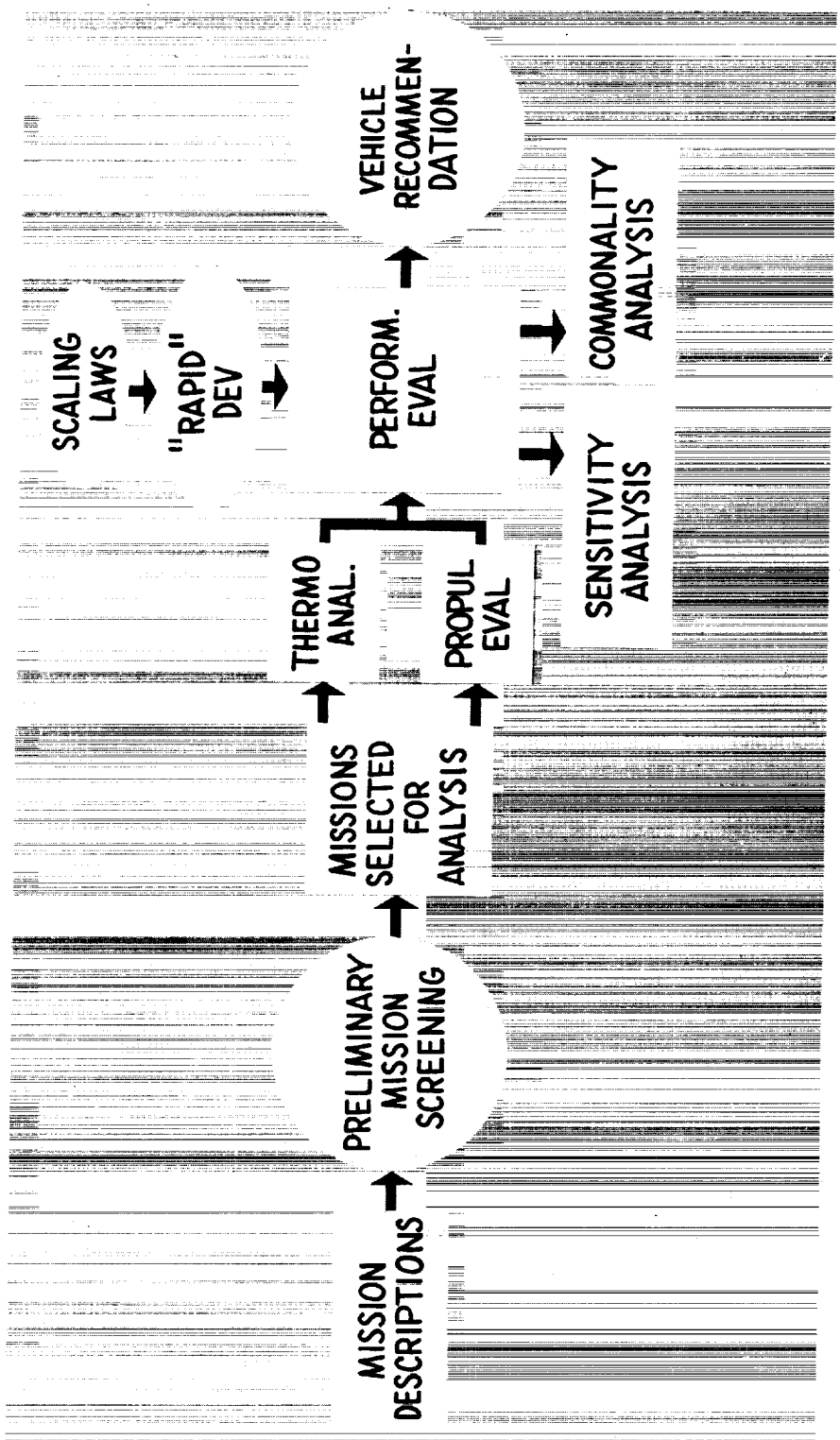

Fig. 2 Mission Evaluation Procedure

Table 2

SELECTED MISSIONS AND MISSION PARAMETERS FOR PRELIMINARY SCREENING

Mission	Payload (lb)	ΔV (ft/sec)	Trip Time (days)	Thrust (lb)
Saturn Unmanned Orbiter	2,000	6,000	1,450	2,000
Jupiter Unmanned Orbiter	2,000	7,600	650	2,000
Venus Manned Flyby – Orbiter Probe	1,500	13,000	115	4,000
Mars Manned Flyby – Orbiter Probe	1,000	21,000	150	4,000
Mars Manned Lander – MEM Ascent Stage	5,260	16,000	220	30,000
Venus Unmanned Orbiter	7,000	13,500	140	8,000
Mars Unmanned Orbiter	8,143	6,950	195	8,000
Lunar Manned Surface Station – Return Stage	19,340	9,186	178	15,000
Earth Manned Synchronous Orbiter – Descent Stage	13,000	9,750	120	20,000
Mars Manned Lander – Planet Departure Stage	92,000	16,000	220/300	100,000
Venus Manned Orbiter – Planet Departure Stage	92,000	14,000	173	100,000
Mars Manned Lander – Earth Departure Stage	110,000	12,900	60	100,000

Table 3

PROPULSION ASSUMPTIONS FOR PRELIMINARY SCREENING

Parameter	Propellent				
	F_2/H_2	O_2/H_2	N_2O_4/A-50	FLOX/CH_4	F_2/NH_3
I_{sp} (sec)	461	446	310	405	407
Mixture Ratio (O/F)	9:1	5:1	1.6:1	5.75:1	3.2:1

Table 4

PROPELLANT PERFORMANCE BY MISSION – STAGE WEIGHT COMPARED WITH N_2O_4/A-50 STAGE

Mission	Stage Weight (% of N_2O_4/A-50)			
	F_2/H_2	O_2/H_2	FLOX/CH_4	F_2/NH_3
Saturn Unmanned Orbiter	95	107	80	79
Jupiter Unmanned Orbiter	81	92	76	75
Venus Manned Flyby – Orbiter Probe	63	71	67	66
Mars Manned Flyby – Orbiter Probe	50	58	55	54
Mars Manned Lander – MEM Ascent Stage	56	65	62	60
Venus Unmanned Orbiter	59	66	64	63
Mars Unmanned Orbiter	70	75	73	73
Lunar Manned Surface Station – Return Stage	66	73	71	69
Earth Manned Synchronous Orbiter – Descent Stage	62	68	68	67
Mars Manned Lander – Planet Departure Stage	53	59	61	60
Venus Manned Orbiter – Planet Departure Stage	54	60	62	61
Mars Manned Lander – Earth Departure Stage	55	60	64	63

missions to Saturn and Jupiter. The validity of the scaling law analysis as a tool for preliminary evaluations was borne out by later detailed analysis.

3.2 SELECTED MISSIONS, STAGES, AND PROPELLANTS

At the completion of the mission analysis task, the four stages listed in Table 5 were recommended for further analysis. The NASA Management Committee selected the Unmanned Mars Orbiter–Orbit Injection Stage as defined in the TRW Voyager studies and the Mars Excursion Module-Ascent Stage as defined by North American-Rockwell Corporation as the two stages to be evaluated in detail during Phase I. These two missions and stages are described in the following paragraphs.

Table 5

STAGES RECOMMENDED FOR ANALYSIS

Stage	Thrust (lb)	Payload (lb)	ΔV (ft/sec)	Duration (days)
Unmanned Saturn Orbiter – Orbit Injection	2,000	2,000	6,000	1,450
Unmanned Mars Orbiter – Orbit Injection	8,000	8,143	6,950	195
Lunar Manned Station – Return Stage	15,000	19,340	9,186	178
Mars Excursion Module – Ascent Stage	30,000	5,260	16,000	220

3.2.1 Baseline Mars Orbiter

The baseline Phase I Mars Orbiter has an 8,000-lb-thrust propulsion system used to insert a spacecraft into an eccentric orbit about Mars. This vehicle is shown in Fig. 3. Nominal parameters for the mission and stage are as follows:

- Payload: 8,143 lb
- Mission Duration: 205 days
- ΔV Total: 6,950 ft/sec
  - 1st Midcourse: 164 ft/sec at 2 days
  - 2nd Midcourse: 164 ft/sec at 165 days
  - Orbit Insertion: 6,294 ft/sec at 195 days
  - Orbit Trim 328 ft/sec at 205 days
- Three-axis stabilization with propulsion system facing the sun

3.2.2 Baseline Mars Excursion Module

The baseline MEM ascent stage has a 30,000-lb-thrust pump-fed propulsion system used to return a four-man capsule from the surface of Mars to a 500-km circular

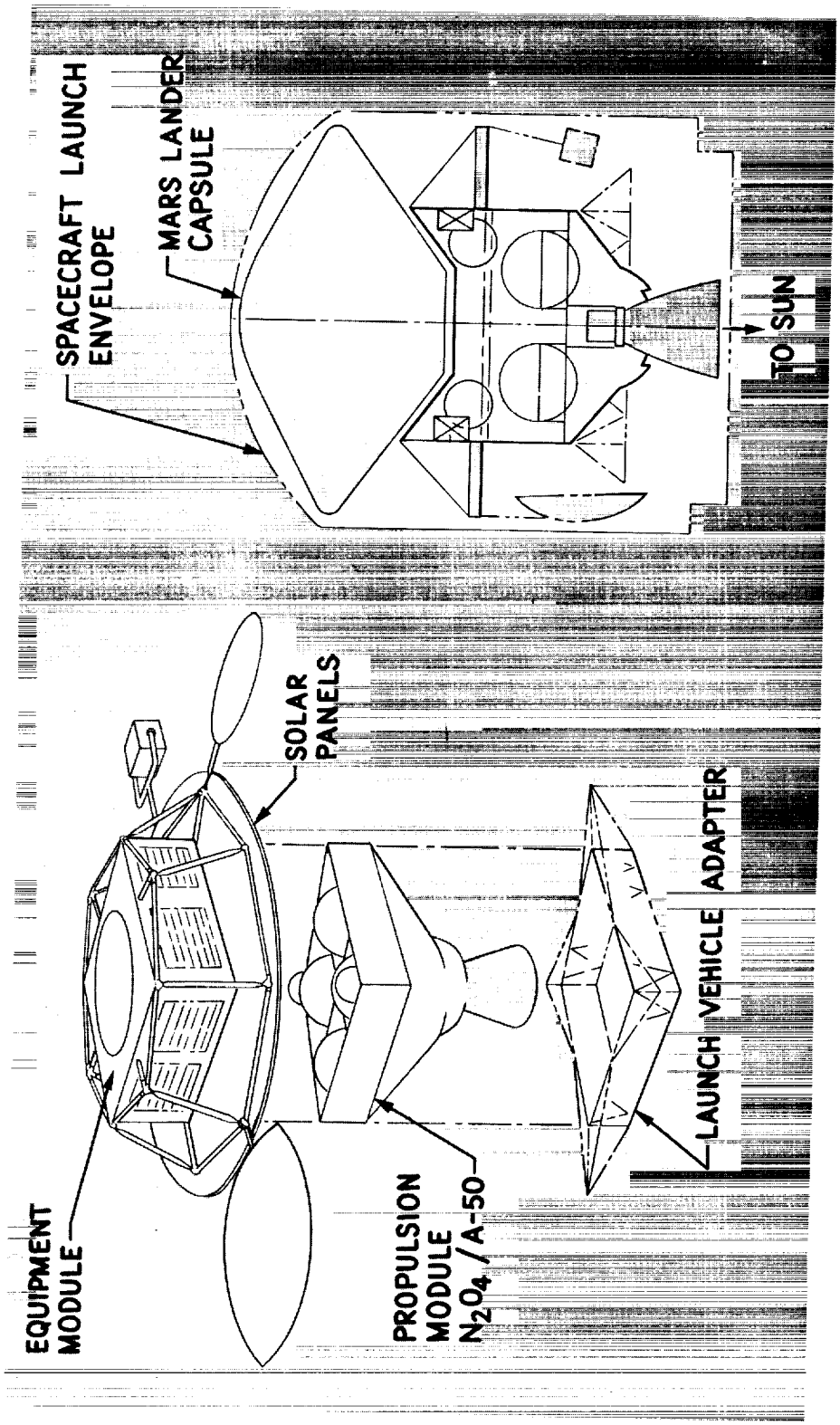

Fig. 3 Baseline Orbiter – TRW Voyager

orbit. This vehicle is shown in Fig. 4. Nominal parameters for this mission and stages are:

- Payload: 5,260 lb
- Mission Duration: 220 days
  - Earth orbit: 30 days
  - Enroute to Mars: 160 days
  - Mars surface: 30 days
- ΔV Total: 16,000 ft/sec*

3.2.3 Propulsion Data Selection

Propellant and propulsion systems data were obtained from the supporting engine companies for point designs with systems of 8,000 lb and 30,000 lb thrust. These thrust levels were specified in the Mars Orbiter and MEM Ascent Stage studies, respectively. These data were reviewed and selections made by Lockheed for use in Phase I analyses.

The nominal specific impulse values selected are considered to be realistically optimistic for the 1975 time period. Since the degree of optimism is believed to be uniform across the matrix of propellants, the comparison of one propellant with any other was not affected.

Propellant and propulsion systems characteristics assumed for the 8,000-lb-thrust Mars Orbiter stage and for the 30,000-lb-thrust Mars Excursion Module Ascent Stage are presented in Tables 6 and 7, respectively.

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*16,000 ft/sec is the nominal mission, not the basic design mission assumed by North American-Rockwell.

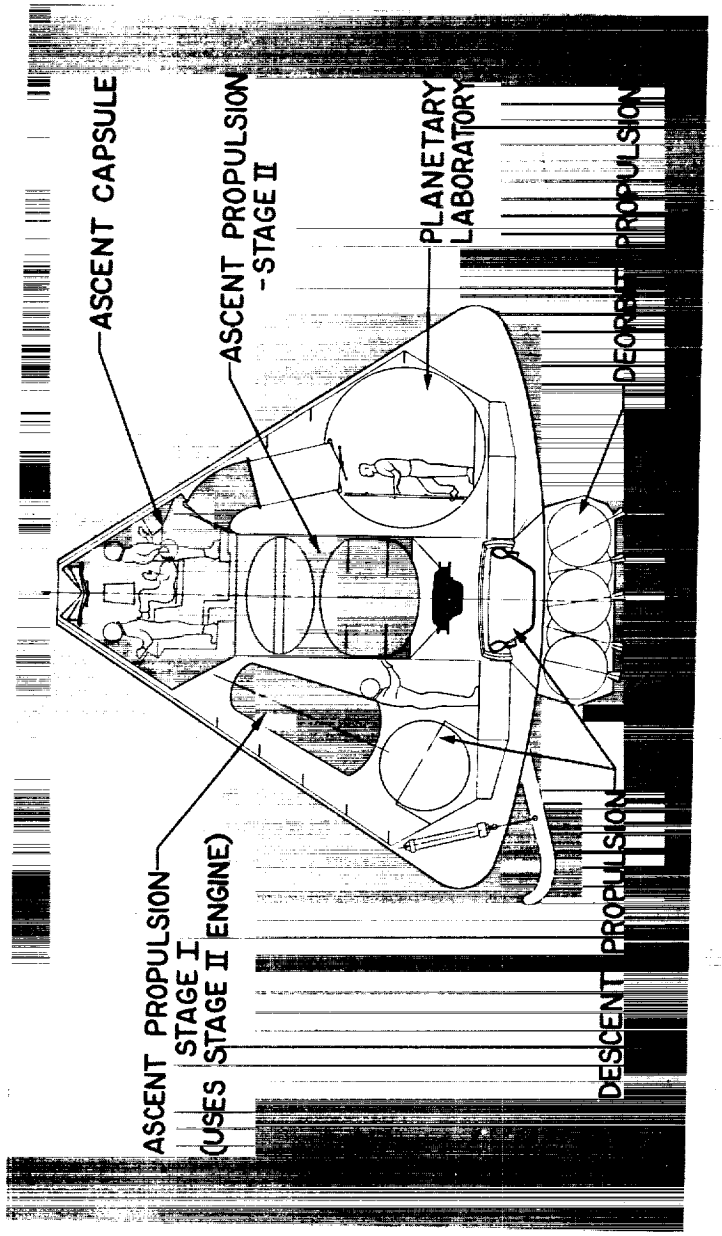


Fig. 4 Baseline Mars Excursion Module

Table 6

MARS ORBITER PROPULSION SYSTEM CHARACTERISTICS
(Bell Nozzle 8,000-lb Thrust)

Propellant	Mixture Ratio (O/F)	Pump-Fed Systems			Pressure-Fed Systems			Cooling
		Chamber Pressure (psia)	I_{sp} (a) (sec)	Engine Wt (lb)	Chamber Pressure (psia)	I_{sp} (a) (sec)	Engine Wt (lb)	
F_2/H_2	13	900	468	173 (b)	100	442	375	Regenerative
O_2/H_2	6	900	451	173 (b)	100	445	380	Regenerative
$FLOX/CH_4$	5.75	600	410	152	100	387	375	Regenerative
OF_2/CH_4	5.3	600	410	152	100	396	380	Regenerative
OF_2/B_2H_6	3.82	-	-	-	100	414	384	Ablative
F_2/NH_3	3.3	1,500	408	192	100	386	375	Regenerative
N_2O_4/A-50	2.0	750	335	158	100	328	330	Ablative
ClF_5/MHF-5	2.4	750	342	167	100	330	384	Ablative

(a) Based on $\epsilon = 100$
(b) Extendable bell nozzle

Table 7

MEM PROPULSION SYSTEM CHARACTERISTICS
(AEROSPIKE NOZZLE – 30,000-LB THRUST)

Propellant	Mixture Ratio (O/F)	Chamber Pressure (psia)	Expansion Ratio ϵ	I_{sp} (sec)	Engine Weight (lb)	Cooling
F_2/H_2	13	750	75	463	440	Regenerative
O_2/H_2	6	750	100	449	520	Regenerative
$FLOX/CH_4$	5.7	750	75	400	440	Regenerative
OF_2/CH_4	5.3	750	75	406	460	Regenerative
F_2/NH_3	3.3	750	75	397	440	Regenerative
ClF_5/MHF-5	2.4	750	100	336	475	Ablative

3.2.4 Propellants

The propellants of interest can be grouped into cryogens, space storables, and earth storables. The cryogens, using liquid hydrogen as fuel, are characterized by low bulk density, high specific impulse, and nonoverlapping liquidus range. Earth storables are in the liquid state at earth ambient temperatures, and are characterized by high bulk density, low specific impulse, and overlapping liquidus range. The space storables exhibit moderately high specific impulse, high bulk density, and may or may not have an overlapping liquidus range.

The propellants selected for the vehicle analysis are shown in Fig. 5, together with the liquidus temperature range of each. The propellant OF_2/B_2H_6 was not analyzed for the MEM because only pump-fed systems were considered, and OF_2/B_2H_6 was not recommended for pump-fed application. N_2O_4/A-50 was also not considered since the MEM was basically a $FLOX/CH_4$ system, and only the best performing earth storable, ClF_5/MHF-5, was used for comparison. For the Mars Orbiter vehicle, all of the propellants were evaluated for pressure-fed systems, and all but OF_2/B_2H_6 were evaluated for the pump-fed systems.

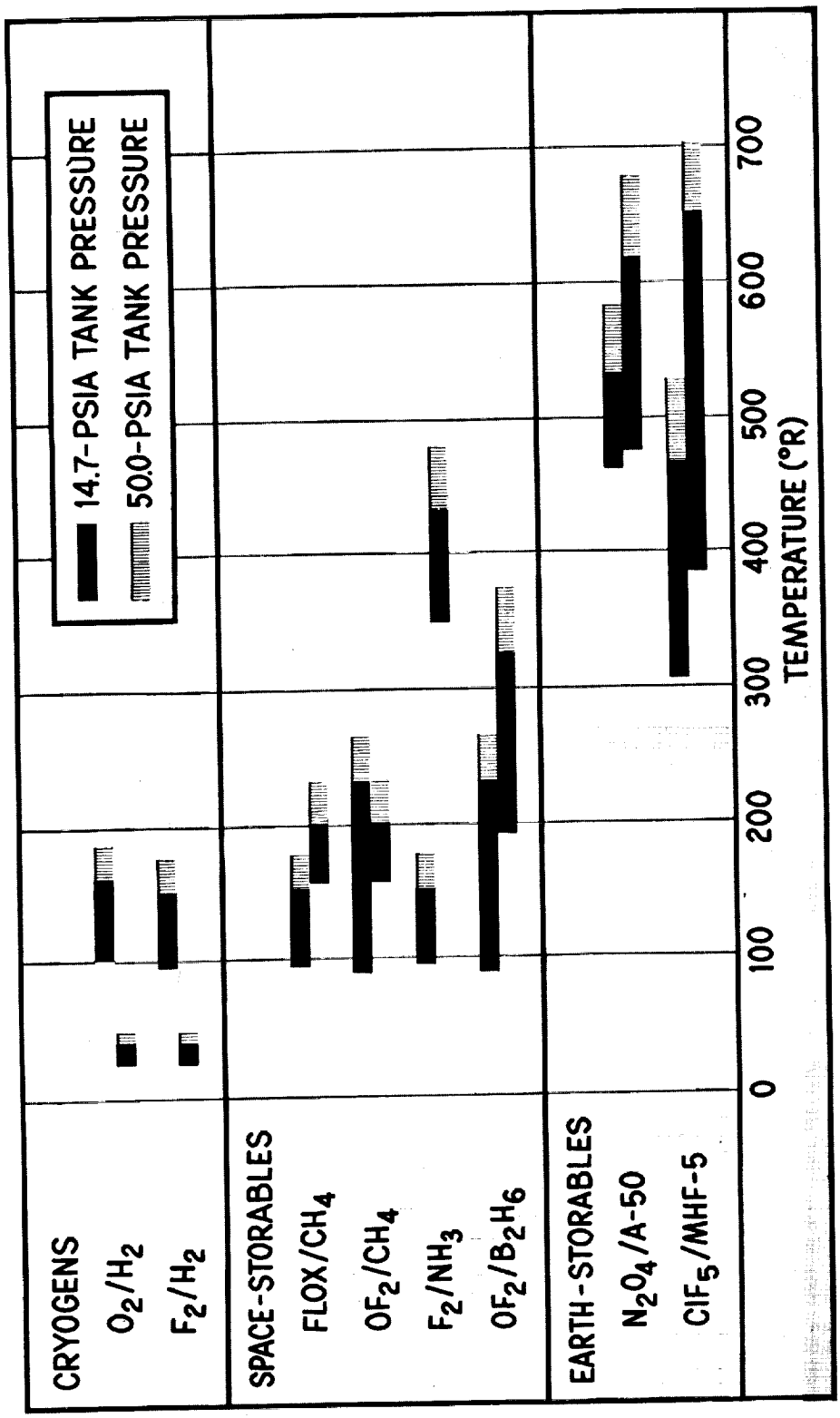


Fig. 5 Propellant Liquid Temperature Range

3.3 PERFORMANCE COMPARISON – MEM AND MARS ORBITER

A performance comparison of the Phase I Mars Orbiter and MEM Ascent Stages is presented in the following paragraphs. Configuration drawings may be found in Ref. 1.

Mars Orbiter. The basic Mars Orbiter propulsion stage, using the LM descent engine burning N_2O_4/A-50, was estimated by TRW Systems to weigh 13,453 lb to orbit a payload of 8,143 lb. Using the same propellant and basic stage design, the system was revised to use a new, pump-fed N_2O_4/A-50 engine with delivered specific impulse increased from 305 to 335 sec. The resulting stage weight was reduced from 13,453 to 9,535 lb to form a new "baseline" orbiter. Figure 6 shows the stage weights for all propellant combinations, with non-vented, sun oriented tanks, for both pump-fed and pressure-fed systems. The pump-fed systems outperform the pressure-fed systems by from 11 percent for F_2/NH_3 to 21 percent for F_2/H_2. Figure 7 shows the stage weights for an alternate spacecraft orientation and the results of venting H_2. Sun-shading the cryogens and space storables improved performance about 7 percent and 2 percent, respectively. Earth storables were best when not shaded. Venting H_2 improved performance about 5 percent. The space-storable and cryogenic stages outperform the N_2O_4/A-50 stages in all cases by inserting the fixed 8,143-lb payload with a lighter propulsion stage. The percent reduction in stage weight over that of the new "baseline" N_2O_4/A-50 stages is presented in Table 8 for each propellant combination, feed system, and spacecraft orientation. The lightest pump-fed stage is achieved with F_2/H_2, and the lightest pressure-fed stage with OF_2/B_2H_6.

Table 8

PERCENT REDUCTION IN MARS ORBITER STAGE WEIGHT OVER N_2O_4/A-50

Orientation	Propellant						
	F_2/H_2	O_2/H_2	FLOX/CH_4	OF_2/CH_4	F_2/NH_3	OF_2/B_2H_6	ClF_5/MHF-5
Pump-Fed (Sun on Tanks)	24	11	16	17	16	–	3
Pump-Fed (Tanks in shade)	29	18	19	19	18	–	3
Pressure-Fed (Sun on Tanks)	12	4	11	14	14	20	2

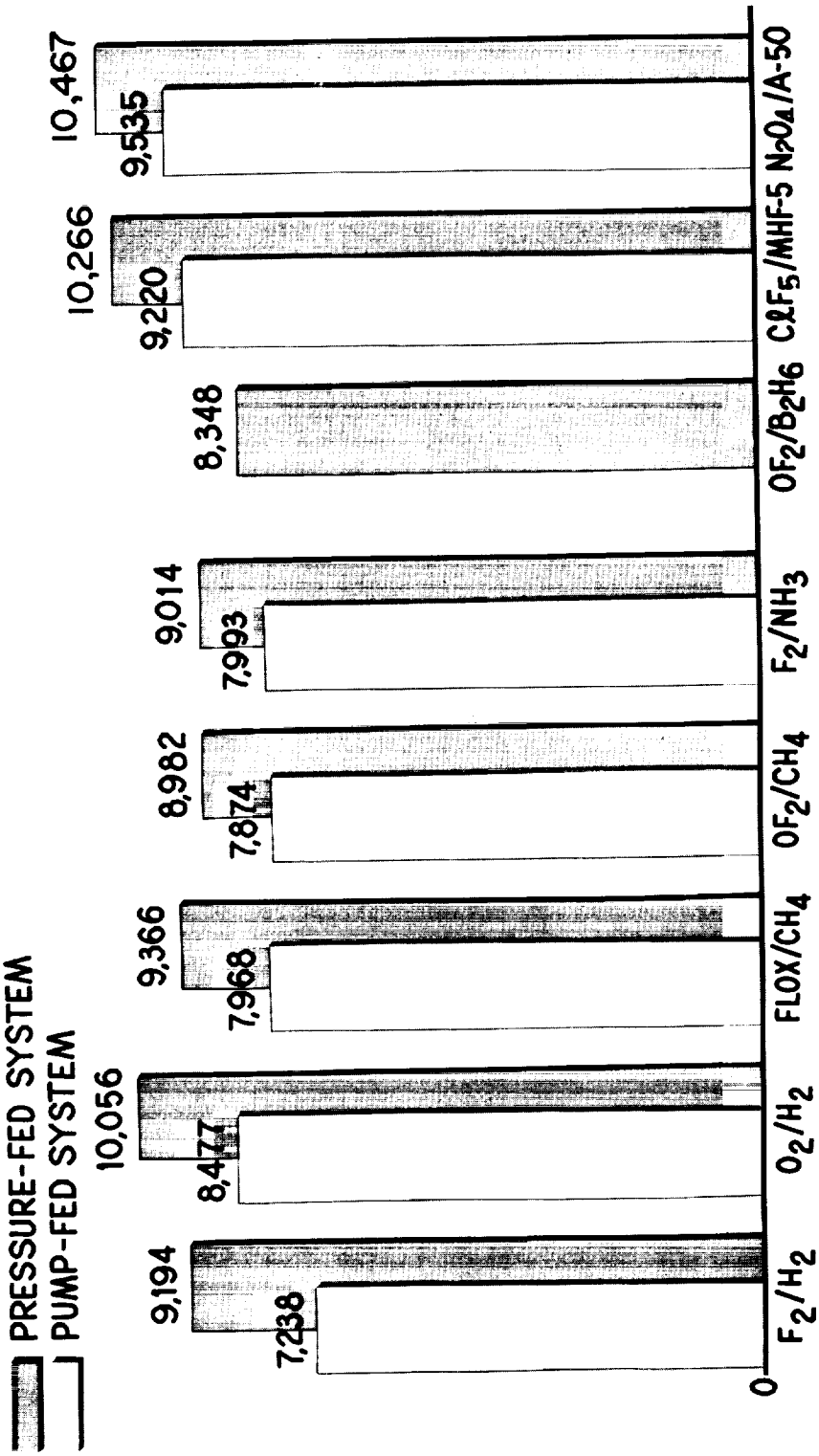


Fig. 6 Mars Orbiter Propulsion Stage Weights – Pressure-Fed and Pump-Fed Systems Comparison Sun-Oriented Tanks

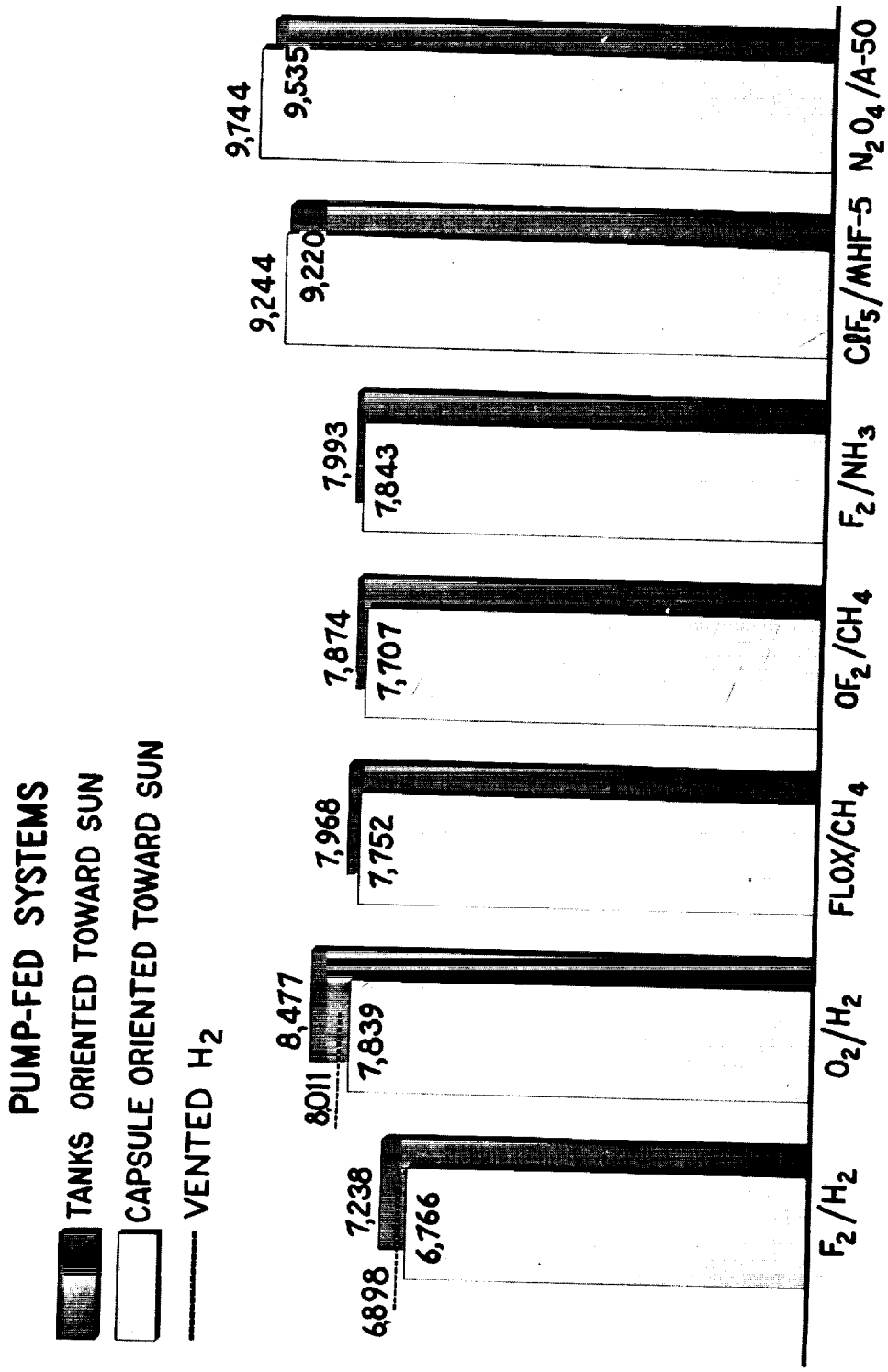


Fig. 7 Mars Orbiter Propulsion Stage Weights – Effects of Orientation and Venting

MEM Ascent Stage. The basic MEM Ascent Stage, burning FLOX/CH_4, was estimated by North American-Rockwell to weigh 24,600 lb in order to launch and orbit a payload of 5,250 lb, providing a total ΔV of 16,000 ft/sec. Using the same propellant with a modified stage design, the MEM was revised to a nominal diameter of 30 ft, spherical propellant tanks, and an increase in specific impulse from 383 to 400 sec. The resulting ascent stage weight was reduced from 24,600 lb to 21,301 lb to form a new "baseline." Figure 8 shows the stage weights for all propellant combinations (except O_2/H_2 which exceeded the volume limits).

The percent change in the weight over that of the new "baseline" FLOX/CH_4 stage is presented in Table 9 for each propellant combination. The lightest weight stage is achieved with F_2/H_2.

Table 9

PERCENT CHANGE IN MEM ASCENT STAGE WEIGHT FROM FLOX/CH_4

Propellant	F_2/H_2	OF_2/CH_4	F_2/NH_3	ClF_5/MHF-5
Percent Change	-11	-3	-1	+32

3.4 SENSITIVITY ANALYSES

During Phase I an assessment was made to determine the system effects of varying several of the design parameters for the Mars Orbiter pump-fed vehicle. The parameters varied were mission length, surface coatings, meteoroid flux, specific impulse, insulation conductivity, propellant initial condition, vent versus nonvent, vehicle orientation to the sun, and a "worst-on-worst" combination. The results of this assessment were reported in detail in Ref. 1, and only two items, propellant initial condition and vehicle orientation, are discussed here.

3.4.1 Propellant Initial Condition

Analysis was made of the effect of the initial condition of hydrogen and the venting of hydrogen for the cryogenic systems. The comparison was made with the sun-on-tank

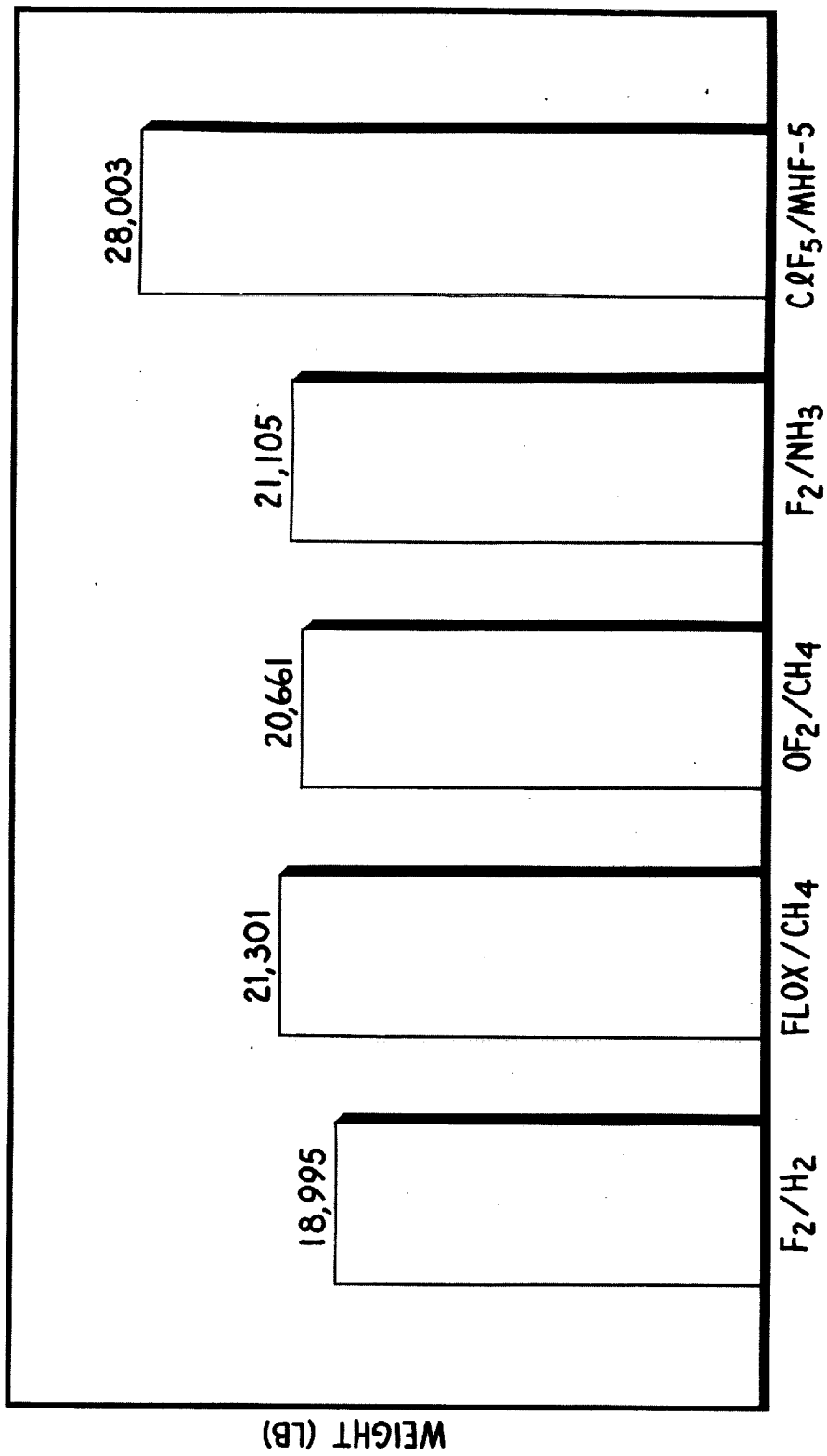


Fig. 8 MEM Ascent Stage Weights

orientation comparing (1) saturated, (2) triple-point, (3) 50-percent slush hydrogen, and (4) venting the hydrogen. With the vented hydrogen case, the oxidizer is cooled by passing the vented hydrogen through a heat exchanger in the oxidizer tank.

Table 10 lists the operating pressure, insulation thickness, and propulsion module weights for the cases studied. The percent weight reductions from the nominal for F_2/H_2 are: Triple point 1.5 percent, slush 2.4 percent, and vented 4.7 percent. The corresponding reductions for O_2/H_2 are: 3.0 percent, 4.0 percent, and 5.5 percent. Combinations of triple-point or slush with venting were not examined. Note, in the following subsection, that shading the tanks from the sun is more effective than subcooling, slush, or venting.

3.4.2 Vehicle Orientation

The effect of orienting the vehicle so that its propellant tanks are exposed to the sun or shielded from the sun provided the most significant effect in terms of insulation thickness, operating pressure, and system weight. Table 11 presents these data. It is significant that the hydrogen tank pressure can be reduced from 130 to 80 psi, the insulation thickness reduced from 4-5/8 to 1-3/4 in., and the system weight for this F_2/H_2 propulsion module reduced from 7,238 to 6,766 lb for a 7-percent weight saving by orienting the vehicle so that the propellant tanks are shaded. This effect varies from about 2 percent for the space storables to 8 percent for O_2/H_2. For the earth storables, sun-facing tanks are desired because sun-shielded orientation would require greater insulation thickness to prevent the propellants from freezing.

3.5 PROPELLANT SELECTION FACTORS

During Phase I an assessment was made of factors that can influence the choice of a propellant. These factors include system performance, volumetric constraints, system sensitivities to environment or to off-optimum operations, system complexity/reliability, development time/availability, cost, compatibility, and commonality. The actual selection of propellants for a specific application will depend on many additional factors peculiar to the requirements and circumstances present in the program. The final choice must rest with the vehicle program office.

Table 10

MARS ORBITER SENSITIVITY TO PROPELLANT INITIAL CONDITION AND VENTING
(Sun-On-Tank, Pump-Fed, 205-Day Mission, Optimum α/ϵ = 0.05/0.80)

PROPELLANT	NONVENTED									VENTED			
	SATURATED H_2			TRIPLE-POINT H_2			50% SLUSH H_2			SATURATED H_2			
	P	T	WT	P	T	WT	P	T	WT	P	T	WT	BOILOFF (LB)
F_2	40	1-1/8		40	1-1/8		40	1-1/8		15	1/2		
			7238			7126			7067			6898	100
H_2	130	4-5/8	(REF)	95	3-3/4	(-1.5%)	82	3-1/2	(-2.4%)	76	2	(-4.7%)	
O_2	58	3/4		58	3/4		58	3/4		15	1/2		
			8477			8226			8141			8011	140
H_2	96	4-5/8	(REF)	66	3-3/4	(-3%)	66	3	(-4.0%)	64	2	(-5.5%)	

P = TANK MAXIMUM OPERATING PRESSURE (PSIA)

T = INSULATION THICKNESS (IN.)

WT = PROPULSION MODULE WEIGHT (LB)

Table 11

MARS ORBITER SENSITIVITY TO ORIENTATION
(Pump-Fed, Nonvented, 205-Day Mission, Optimum α/ϵ)

Propellant	Sun on Tank			Sun on Capsule			Weight Change (%)
	P[a]	T[b]	WT[c]	P	T	WT	
F_2 H_2	40 130	1-1/8 4-5/8	7,238	<15 80	Min 1-3/4	6,766	-7.0
O_2 H_2	58 96	3/4 4-5/8	8,477	<15 72	Min 1-3/4	7,839	-8.1
FLOX CH_4	57 107	1-1/2 3/4	7,968	32 <15	Min Min	7,752	-2.8
OF_2 CH_4	45 107	1-1/8 3/4	7,874	<15 <15	Min Min	7,707	-2.2
F_2 NH_3	59 16	1-3/8 Min	7,993	<15 <15	Min Min	7,843	-1.9
N_2O_4 A-50	<15 <15	Min Min	9,535	<15 <15	2 2	9,744	+2.2
ClF_5 MHF-5	<15 <15	Min Min	9,220	<15 <15	3/8 3/8	9,244	+0.3

(a) Tank maximum operating pressure (psia)
(b) Insulation thickness (in.)
(c) Propulsion module weight (lb)

An approach to use in considering some of the factors of importance in selecting a propellant is outlined in Fig. 9. In following the paths shown, a considerable amount of personal judgment and knowledge will be required.

In Step (1), the requirements placed on the propulsion stage for thrust, ΔV, restarts, mission duration, throttling, minimum impulse bit, launch date, etc., are stated in detail. A decision must then be made at Step (2) as to whether pump or pressure-feed will be specified, recognizing that pump feed will generally result in the lowest weight and most compact system.

In Step (3), the size of the required stage is estimated and the best performing propellant selected for further analysis. In this step, space storables include FLOX/CH_4, FLOX/C_3H_8, OF_2/CH_4, OF_2/C_3H_8, OF_2/MMH, and F_2/NH_3. Each of these propellants was found to be space storable for the missions analyzed with no overriding storage reason found for choosing one over another, although not all of these combinations were studied in detail. The specific impulse for pump-fed applications is comparable (± 1 percent), except for OF_2/C_3H_8, which provides a specific impulse about 2.5 percent below the mean. Pressure-fed data were obtained only for OF_2/CH_4, FLOX/CH_4, and F_2/NH_3, with the OF_2/CH_4 providing about 2.5 percent higher specific impulse than the latter two. OF_2/B_2H_6 is treated separately from the other space storables, since its use is restricted to pressure-fed applications and it provides a higher specific impulse than other pressure-fed space storables.

In Step (4), a check must be made to determine that mission-peculiar requirements, such as need for propellant or total stage sterilization, can be met.

In Step (5), a check is made to ensure that volume and dimensional limits for the spacecraft and launch system are met. Propellant and engine data are provided as an aid in this assessment.

In Step (6), a check is made to ensure that the selected propellant is compatible with spacecraft and payload equipment and personnel. Some general guides are provided.

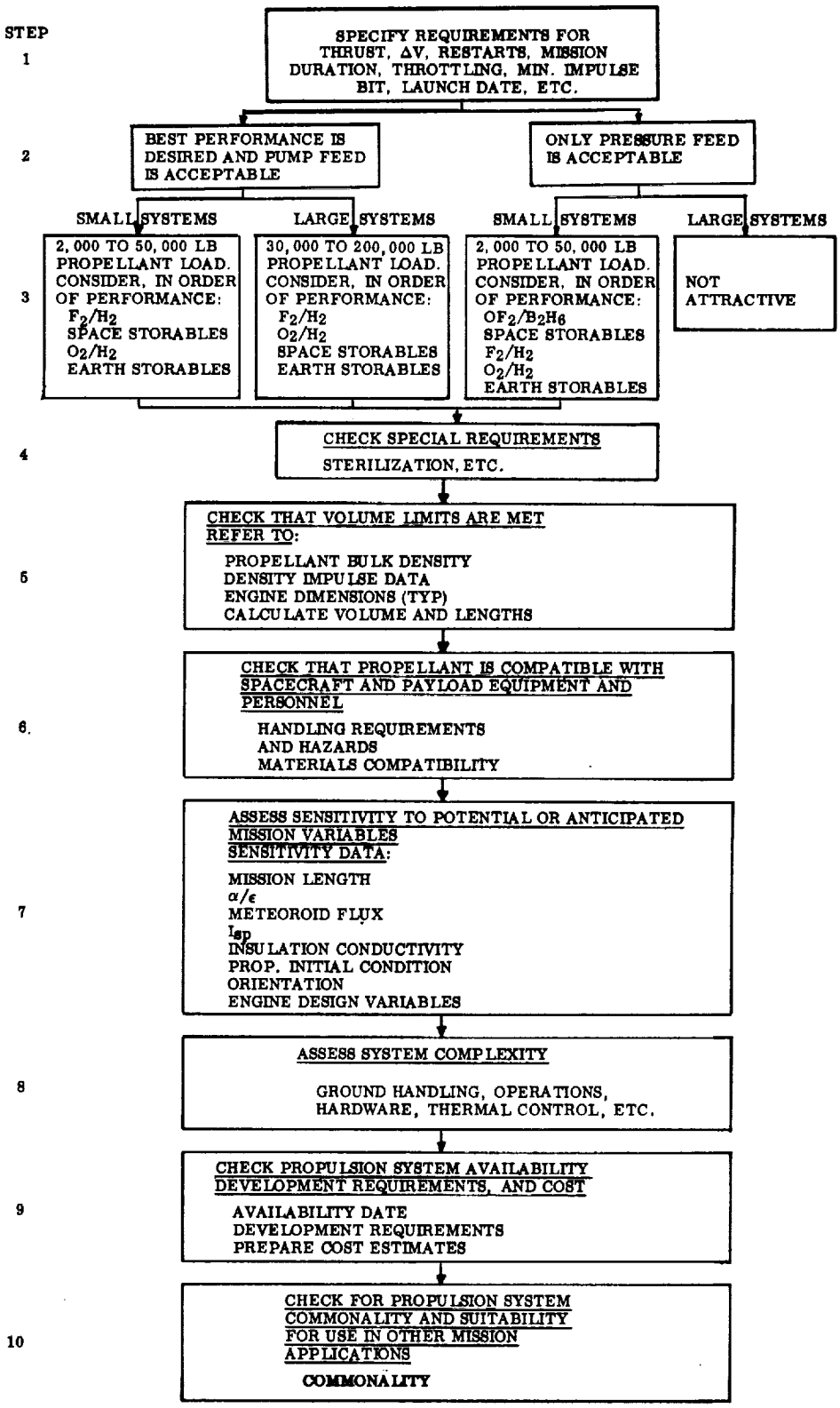

STEP
1
SPECIFY REQUIREMENTS FOR THRUST, ΔV, RESTARTS, MISSION DURATION, THROTTLING, MIN. IMPULSE BIT, LAUNCH DATE, ETC.
2
BEST PERFORMANCE IS DESIRED AND PUMP FEED IS ACCEPTABLE
ONLY PRESSURE FEED IS ACCEPTABLE
SMALL SYSTEMS
LARGE SYSTEMS
SMALL SYSTEMS
LARGE SYSTEMS
3
2,000 TO 50,000 LB PROPELLANT LOAD. CONSIDER, IN ORDER OF PERFORMANCE: F2/H2 SPACE STORABLES O2/H2 EARTH STORABLES
30,000 TO 200,000 LB PROPELLANT LOAD. CONSIDER, IN ORDER OF PERFORMANCE: F2/H2 O2/H2 SPACE STORABLES EARTH STORABLES
2,000 TO 50,000 LB PROPELLANT LOAD. CONSIDER, IN ORDER OF PERFORMANCE: OF2/B2H6 SPACE STORABLES F2/H2 O2/H2 EARTH STORABLES
NOT ATTRACTIVE
4
CHECK SPECIAL REQUIREMENTS
STERILIZATION, ETC.
5
CHECK THAT VOLUME LIMITS ARE MET
REFER TO:
PROPELLANT BULK DENSITY
DENSITY IMPULSE DATA
ENGINE DIMENSIONS (TYP)
CALCULATE VOLUME AND LENGTHS
6.
CHECK THAT PROPELLANT IS COMPATIBLE WITH SPACECRAFT AND PAYLOAD EQUIPMENT AND PERSONNEL
HANDLING REQUIREMENTS AND HAZARDS
MATERIALS COMPATIBILITY
7
ASSESS SENSITIVITY TO POTENTIAL OR ANTICIPATED MISSION VARIABLES
SENSITIVITY DATA:
MISSION LENGTH
α/ε
METEOROID FLUX
Isp
INSULATION CONDUCTIVITY
PROP. INITIAL CONDITION
ORIENTATION
ENGINE DESIGN VARIABLES
8
ASSESS SYSTEM COMPLEXITY
GROUND HANDLING, OPERATIONS, HARDWARE, THERMAL CONTROL, ETC.
9
CHECK PROPULSION SYSTEM AVAILABILITY DEVELOPMENT REQUIREMENTS, AND COST
AVAILABILITY DATE
DEVELOPMENT REQUIREMENTS
PREPARE COST ESTIMATES
10
CHECK FOR PROPULSION SYSTEM COMMONALITY AND SUITABILITY FOR USE IN OTHER MISSION APPLICATIONS
COMMONALITY

Fig. 9 Propellant Selection Factors to be Considered

In Step (7), a check is made of sensitivity to potential or anticipated mission variables, such as spacecraft orientation, venting requirements, meteoroid flux, propellant tank surface coating properties, and specific impulse tolerance. Sensitivity data for a typical mission, a Mars Orbiter, are presented as a guide.

In Step (8), the overall complexity of the selected system is evaluated to assess, on a preliminary basis, the probability of meeting reliability requirements.

In Step (9), the development status and requirements, availability date, and estimated development and procurement costs are assessed.

In Step (10), a check is made for propulsion system commonality and suitability for use in alternate mission applications.

If at any step the propellant under consideration is found unacceptable, or is judged not a good choice through marginal acceptance at one or more steps, the procedure is started again at Step (3), and the next best performer is considered.

Section 4
RESULTS – PHASE II

4.1 PROPELLANT SENSITIVITY ANALYSIS

Under the task heading of Propellant Sensitivity Analysis, six separate subtasks were analyzed as the first items of work in Phase II. These subtasks, Temperature Dependent Insulation K factors, Shadow Shield Effects, Engine Start Mode Effects, Thrust and Chamber Pressure Sensitivity, Propellant Leak Rate Sensitivity, and Complexity Analysis are discussed in turn below.

4.1.1 Temperature Dependent K Factors

A study was conducted to evaluate the peformance effects of using insulation conductivity (K) values based on actual inner and outer boundary temperatures rather than fixed K values. Phase I analyses had been performed assuming fixed K values were:

For H_2 tanks	2.5×10^{-5} BTU/ft-hr-°R
For F_2, O_2, OF_2, FLOX, and CH_4	5.0×10^{-5} BTU/ft-hr-°R
For Earth Storables	10.0×10^{-5} BTU/ft-hr-°R

The pump-fed Mars Orbiter vehicle was assumed to have sun-on-tank orientation because of the greater sensitivity to K factors. F_2/H_2, O_2/H_2, FLOX/CH_4, OF_2/CH_4, and F_2/NH_3 were the propellants investigated, and insulation surface temperatures and resulting K values at Earth and Mars are presented in Table 12.

The net result of the refined conductivity analysis was lower vehicle weights. The changes in tank pressures and insulation thicknesses are small and result in propulsion module weight savings of from 0.3 to 1.5 percent as shown in Table 13.

Table 12

TEMPERATURE DEPENDENT INSULATION CONDUCTIVITY

	Insulation Surface Temperature		Conductivity Btu/Hr-Ft-°R × 10^{-5}			Change in Weight*	
	Earth (°R)	Mars (°R)	Baseline	K(T) Earth	K(T) Mars	Residuals (lb)	Inert (lb)
F_2	302	252	5.0	2.54	2.03	- 5.2	- 3.7
H_2	389	330	2.5	2.66	1.98	- 8.0	-11.5
O_2	302	252	5.0	2.54	2.03	- 6.8	+ 0.2
H_2	389	330	2.5	2.66	1.98	+ 5.0	-45.
FLOX	384	326	5.0	3.58	2.80	- 9.8	- 0.5
CH_4	384	326	5.0	4.08	3.23	- 1.5	- 1.0
OF_2	384	326	5.0	4.44	3.56	- 3.4	+ 0.4
CH_4	384	326	5.0	4.08	3.23	- 4.1	- 0.5
F_2	384	326	5.0	3.56	2.77	- 0.8	- 7.0
NH_3	555	465	10.0	13.2	10.2	0	0

*Relative to baseline

Table 13

INSULATION CONDUCTIVITY PERFORMANCE COMPARISON

Mars Orbiter Vehicle, Pump-Fed, Nonvented, Sun-on-Tank

PROPELLANT	BASELINE			TEMPERATURE DEPENDENT K FACTOR		
	Tank Max. Operating Press. (psia)	Insulation Thickness (in.)	Propulsion Module Weight (lb)	Tank Max. Operating Press. (psia)	Insulation Thickness (in.)	Propulsion Module Weight (lb)
F_2	40	1-1/8	7238	78	1	7173
H_2	130	4-5/8		174	4	
O_2	58	3/4	8477	42	1	8352
H_2	96	4-5/8		100	4-1/4	
FLOX	57	1-1/2	7968	56	1-1/2	7918
CH_4	107	3/4		81	1	
OF_2	45	1-1/8	7874	58	1	7847
CH_4	107	3/4		79	1	
F_2	59	1-3/8	7993	55	1-1/2	7964
NH_3	16	Min.		16	Min.	

Min. = 1/2-in. foam

4.1.2 Shadow Shield Effects

Two basic shadow shield concepts were evaluated for the pump-fed Mars Orbiter propulsion module using F_2/H_2 propellants. One concept featured a single shield mounted aft of sun-facing tanks. The second concept utilized multiple shields mounted forward between the propulsion module and the payload for a spacecraft with payload facing the sun. Figure 10 shows that the shadow shields are quite effective, approaching the minimum system weight achievable with no heating. The forward shield (three shields spaced equally in a one foot extension between propulsion module and payload) is slightly more efficient than the aft shield but has a larger structural penalty. Tank operating pressures, insulation thickness, and propulsion module weights are compared in Table 14.

4.1.3 Engine Start Mode Effects

A comparison was made of alternate engine start modes including idle mode, liquid containment device, and ullage rockets. The results of this comparison are presented in Table 15. From this table it is evident that use of ullage rockets (attitude control system gas jets were assumed in the analysis) rather than idle mode is accomplished with only a very minor weight penalty. Penalties do become appreciable when it is assumed that a containment device is used and that pressurant gas is initially injected into a cold liquid propellant such as hydrogen. It appears evident that the cold propellants should be settled prior to introducing the pressurant.

4.1.4 Thrust and Chamber Pressure Sensitivity

The Mars Orbiter analyzed in Phase I was assumed to use an 8,000 pound thrust engine with a chamber pressure of 100 psi for the pressure-fed cases. It was desired to evaluate the effects of varying thrust and chamber pressure for this pressure-fed application.

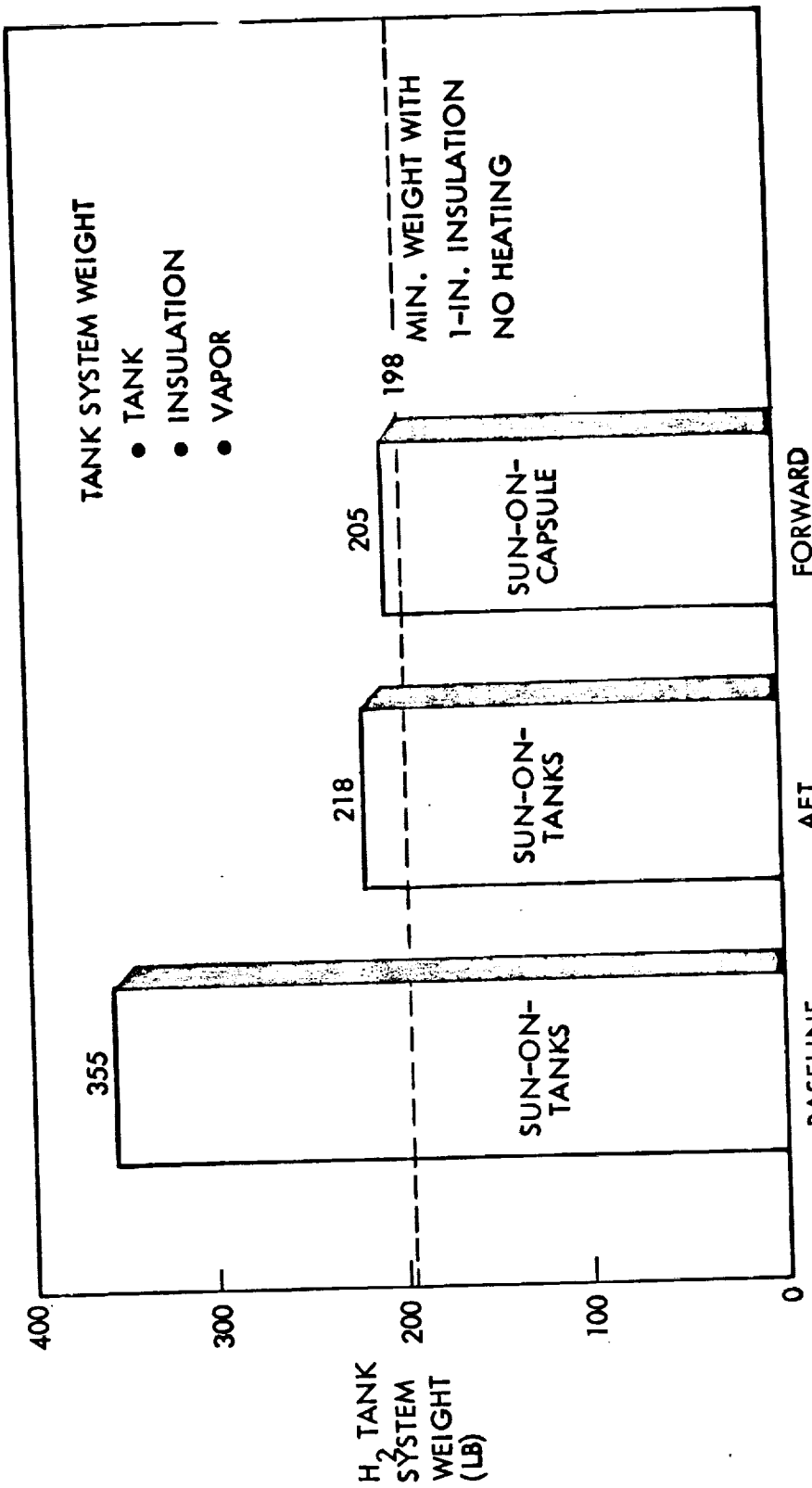


Fig. 10 Shadow Shield Effects

Table 14

SHADOW SHIELD EFFECTS

F_2/H_2 PROPELLANT

CONFIGURATION	TANK OPERATING PRESSURE (PSIA)		INSULATION THICKNESS (IN.)		PROPULSION MODULE WEIGHT (LB)	Δ
	F_2	H_2	F_2	H_2		
BASELINE (NO SHIELD, SUN ON TANK)	52	60	1-1/4	3	6,990	(0)
AFT SHIELD (SUN ON TANK)	60	61	3/4	1	6,760	(-230)
FORWARD SHIELD (SUN ON CAPSULE)	<15	27	1/2	1	6,764	(-226)

Table 15

ENGINE START MODE EFFECTS

CONCEPT	PROPELLANT	TANK PRESSURE BUILDUP	PROPULSION MODULE WEIGHT (LB)	Δ
IDLE-MODE START	F_2/H_2	WITH WARM PRESSURANT GAS	8,779	(0)
	FLOX/CH_4		9,200	(0)
	N_2O_4/A-50		10,471	(0)
CONTAINMENT DEVICE (NO IDLE-MODE START)	F_2/H_2	WITH PRESSURANT GAS AT PROPELLANT SATURATION TEMPERATURE	9,361	(+582)
	FLOX/CH_4		9,241	(+41)
	N_2O_4/A-50		10,470	(-1)
ULLAGE ROCKETS (NO IDLE-MODE START)	F_2/H_2	WITH WARM PRESSURANT GAS	8,785	(+6)
	FLOX/CH_4		9,206	(+6)
	N_2O_4/A-50		10,477	(+6)

In the analysis both thermodynamic and performance effects were assessed for stages using N_2O_4/A-50, FLOX/CH_4, and F_2/H_2. The results showed that optimum thrust was between 4,000 and 6,000 lb for any combination of propellant and chamber pressure. Optimum chamber pressure was 150 psia for FLOX/CH_4 and N_2O_4/A-50 and 100 psia for F_2/H_2. A summary of the analysis results is presented in Fig. 11.

4.1.5 Propellant Leak Rate Sensitivity

An analysis of propellant leakage through tank shutoff valves was made to determine the importance of both propellant loss and heat conduction due to propellant in the feed lines. Conclusions to the analysis were that:

(1) Propellant loss due to leakage is negligible, varying from an estimated 0.5 lb for N_2O_4 to 4.6 lb for H_2 for the total 205 day Mars Orbiter mission.
(2) Feed line heat leaks caused by propellant leakage could increase total stage heat leaks by 2 to 5 percent. This increase in heating would have only a slight effect on system weight.

4.1.6 Complexity Analysis

A systems complexity analysis was performed in order to compare propellants and to develop simpler fluid systems. It was found that with preferential sun orientation, or shadow shielding, and proper design of insulation and tank support struts the fluid systems of Phase I could be simplified. The final conclusion was that fluid systems for space storables can be essentially identical with those for Earth storables, with propellant remaining in the feed lines during the mission and valves required only at the engine. For H_2 systems an additional propellant isolation valve is required at the propellant tank.

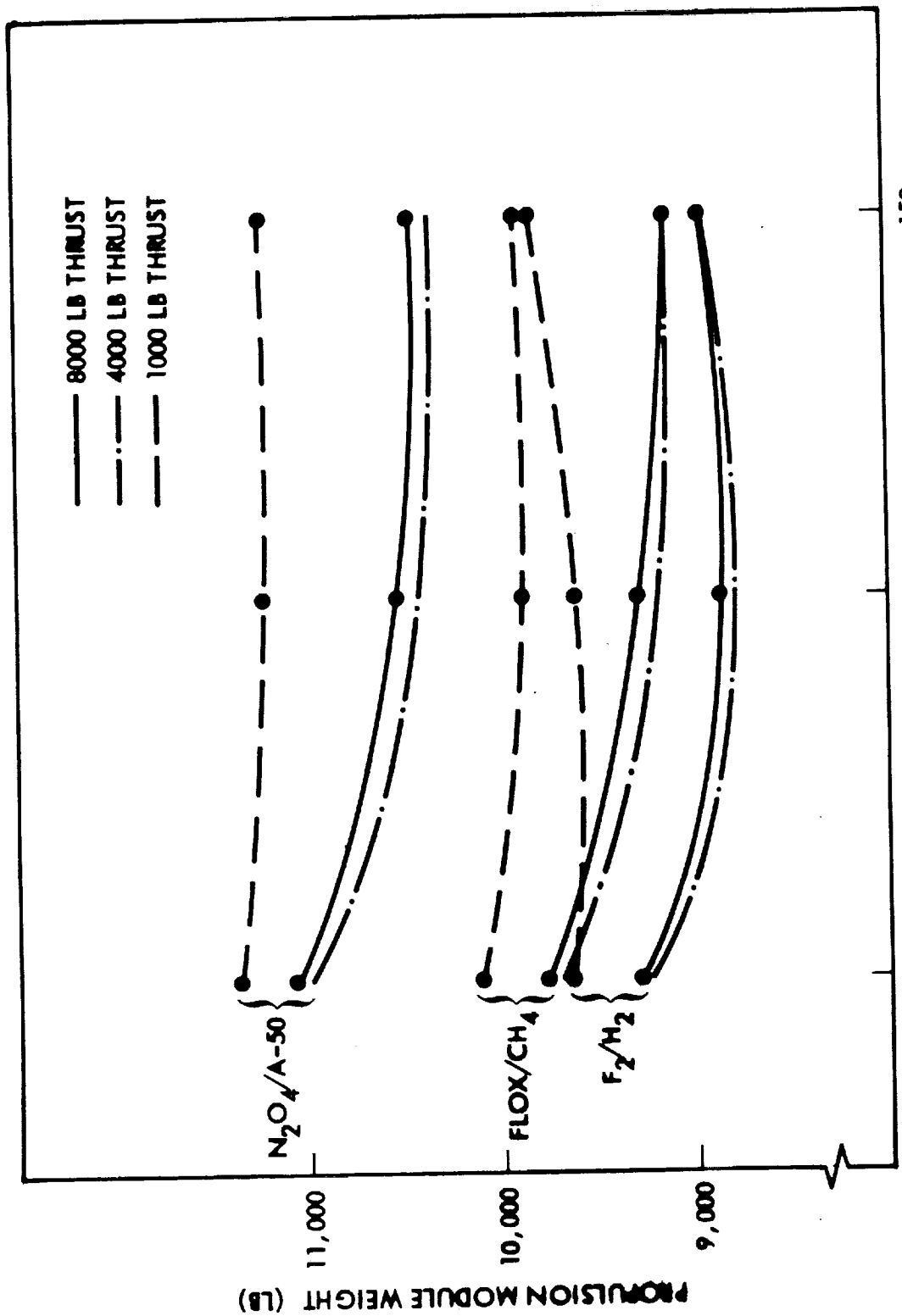


Fig. 11 Thrust and P_c Sensitivity Summary

4.2 PROPULSION ALTERNATIVES FOR MINOR ΔV

4.2.1 Task Objectives and Basis

During the Phase I analysis of a Mars Orbiter propulsion stage, it was assumed that the main engine, throttled from 8,000 lbf to 1,000 lbf, was used to accomplish all secondary burns for midcourse and orbit trim corrections. In Phase II it was desired to compare alternate propulsion modes for meeting these minor ΔV requirements.

The basis for the analysis was as follows:

Mission	Mars Orbiter used in Phase I
Main Engine	8,000 lb thrust, pump fed
Propellants	(1) F_2/H_2, FLOX/CH_4 and N_2O_4/A-50 with common propellant for primary and secondary systems, or,
	(2) N_2O_4/A-50 secondary regardless of propellant in the primary
Maneuvers	(1) 164 ft/sec at 3 days from launch
	(2) 164 ft/sec at 165 days from launch
	(3) 6,294 ft/sec at 195 days from launch
	(4) 328 ft/sec at 205 days from launch
Payload	8,143 lb to Mars orbit
	3,143 lb for final burn (having separated a 5000-lb lander capsule)

The secondary propulsion modes considered are shown in Table 16.

Four propellant combinations, F_2/H_2, FLOX/CH_4, OF_2/B_2H_6, and N_2O_4/A-50, were considered. The secondary propellant for Mode F was cold N_2 gas.

The objective of the analysis was to compare both stage weights and system complexity between propulsion modes. System complexity was determined by comparing the number of critical components and operations for each mode, as described later.

Table 16

SECONDARY PROPULSION MODES

MODE	DESCRIPTION
A	Primary engine throttled (8,000 lbf to 1,000 lbf)
B	Common propellant with primary, separate tanks, two 100 lbf secondary engines
C	Common propellant with primary, common tanks, two 100 lbf secondary engines
D	N_2O_4/A-50 secondary propellant, separate tanks, two 100 lbf secondary engines
E	Primary engine at tank-head-idle
F	Attitude Control System

4.2.2 Analysis Approach

The analysis approach used was to:

1. Set up specific propulsion systems requirements and define engine systems in terms of size, weight, and performance.
2. Analyze pressurization system and thermodynamic effects of a single-burn main propulsion system to determine the reduction in primary system weight and complexity over the baseline multiburn, throttled system.
3. Describe the operational sequences and functional parts of competing systems in order to assess relative complexity.
4. Calculate the performance of competing systems to determine overall stage weights.
5. Discuss factors relating to secondary propulsion mode selection, based on results of performance and complexity analysis.

4.2.3 Propulsion System Requirements

The primary and secondary propulsion systems must satisfy the requirements shown in Table 17. The main engine provides 8,000 lb of thrust for the orbit insertion burn in all cases, and may or may not be throttleable to 1,000-lb thrust for the secondary ΔV requirements. The throttleable main engine was described during Phase I of this contract. If a new secondary system is provided, it is assumed to be pressure-fed and to have two gimbaling, radiation-cooled engines providing 100-lb thrust each.

Table 17

PROPULSION REQUIREMENTS

BURN NO.	TIME AFTER LAUNCH (DAYS)	ΔV (FT/SEC)	APPROXIMATE SPACECRAFT WEIGHT (LB)	MAX IMPULSE LB-SEC
1	3	164	16,300	83,000
2	165	164	16,030	82,800
3	195	6,294	15,760	3,075,000
4	205	328	4,800	49,000

MINIMUM IMPULSE BIT $\simeq$ 1500 LBF-SEC.
IMPULSE VARIATION LIMIT $\simeq \pm 150$ LBF-SEC.

4.2.4 System Performance

System performance is evaluated by comparing the total weight for the baseline stage using a multi-burn main engine with total weights for stages using alternate secondary propulsion systems. In each case the lightest weight system is the baseline, Mode A, wherein the main engine, at full thrust or throttled, is used for all ΔV maneuvers. The next lightest system is Mode B using a secondary system with separate engines, separate tanks, and the same propellant as the primary. Next lightest is Mode D followed in order by Mode C and Mode F.

The use of idle mode for secondary propulsion, designated Mode E, would result in no saving in fixed system weight and would introduce a penalty in reduced I_{sp} over throttled mode. In addition, idle mode is not at present sufficiently developed to assure satisfactory feed and cooling for long burn times. Mode E is therefore not considered further.

The weight penalties for using Mode F, assuming the attitude control system is cold N_2 gas, would be prohibitive and this mode is not considered further. No study was made of hot gas ACS systems, but it appears the system performance could not be better than for Mode D.

Weight penalties for modes A through D are presented in Table 18 for each propellant combination analyzed.

4.2.5 Systems Complexity

An evaluation of systems complexity was made for modes A through D by preparing fluid systems schematics for each mode and propellant combination and then counting critical components and critical operations.

A critical component is defined as one wherein a single failure would cause the mission to be partially or totally lost. Critical components include propellant and pressurant valves, pressure regulators, pumps, and gimbal actuators, but do not include redundant check valves, relief valves, or burst discs. An operational cycle for a component, assumed to consist of once "on" at start of burn and once "off" at the completion of burn is considered as one critical operation. Cycling during a burn, such as occurs for pumps and pressure regulators, is not counted.

The complexity of the working parts of the engines, beyond the fuel and oxidizer valves, was not treated in the analysis and the only engine weight adjustment was for ablative material for the N_2O_4/A-50 primary system.

Table 18

PROPULSION MODE ANALYSIS RESULTS SUMMARY

Propellants / Secondary (1) Propulsion Mode		A	B	C	D (2)	Modes Rejected E	Modes Rejected F (3)
N_2O_4/A-50	Stage Weight (lb)	17,678	17,881	17,925	17,881	Feed and Cooling Not Adequately Developed for Long Burn	Cold Gas Thrust & I_{sp} Too Low
	Weight Penalty (lb)	0	203	247	203		
	Complexity	6th	1st (4)	9th	1st (4)		
FLOX/CH_4	Stage Weight (lb)	16,111	16,179	16,337	16,333	Feed and Cooling Not Adequately Developed for Long Burn	Cold Gas Thrust & I_{sp} Too Low
	Weight Penalty (lb)	0	68	226	222		
	Complexity	7th	3rd	10th	2nd		
F_2/H_2	Stage Weight (lb)	15,433	15,505	15,683	15,692	Feed and Cooling Not Adequately Developed for Long Burn	Cold Gas Thrust & I_{sp} Too Low
	Weight Penalty (lb)	0	72	250	259		
	Complexity	8th	5th	11th	4th		

(1) Modes are defined in Table 16
(2) N_2O_4/A-50 Secondary Propellant
(3) ACS Requirements for a Mars Orbiter are easily met with a small, cold gas system
(4) N_2O_4/A-50 Systems for Modes B and D are identical.
(5) Primary is pump fed. Results may differ for a pressure fed primary.

Table 19 summarizes the numbers of critical components and critical operations for modes A through D.

Table 19

SUMMARY OF CRITICAL COMPONENTS AND CRITICAL OPERATIONS

Propulsion Mode	Propellants (Main & Secondary)	Number of Critical Components	Number of Critical Operations
A	N_2O_4/A-50 & none	10	40
	FLOX/CH_4 & none	10	40
	F_2/H_2 & none	11	44
B	N_2O_4/A-50 & N_2O_4/A-50	17	37
	FLOX/CH_4 & FLOX/CH_4	17	37
	F_2/H_2 & F_2/H_2	19	39
C	N_2O_4/A-50 & N_2O_4/A-50	25	55
	FLOX/CH_4 & FLOX/CH_4	25	55
	F_2/H_2 & F_2/H_2	26	56
D	N_2O_4/A-50 & N_2O_4/A-50	17	37
	FLOX/CH_4 & N_2O_4/A-50	17	37
	F_2/H_2 & N_2O_4/A-50	19	39

4.2.6 Performance and Complexity Summary

A summary of propulsion mode analysis results is presented in Table 18, including total propulsion stage weight, weight penalty, and complexity ranking, by mode and primary propellant. When complexity count is identical between two propellant choices, the warmer propellant is assumed to be less complex.

Both weight penalty and complexity count are plotted on Fig. 12. Here it is evident that the lightest weight system is with throttled main engine. Complexity is decreased

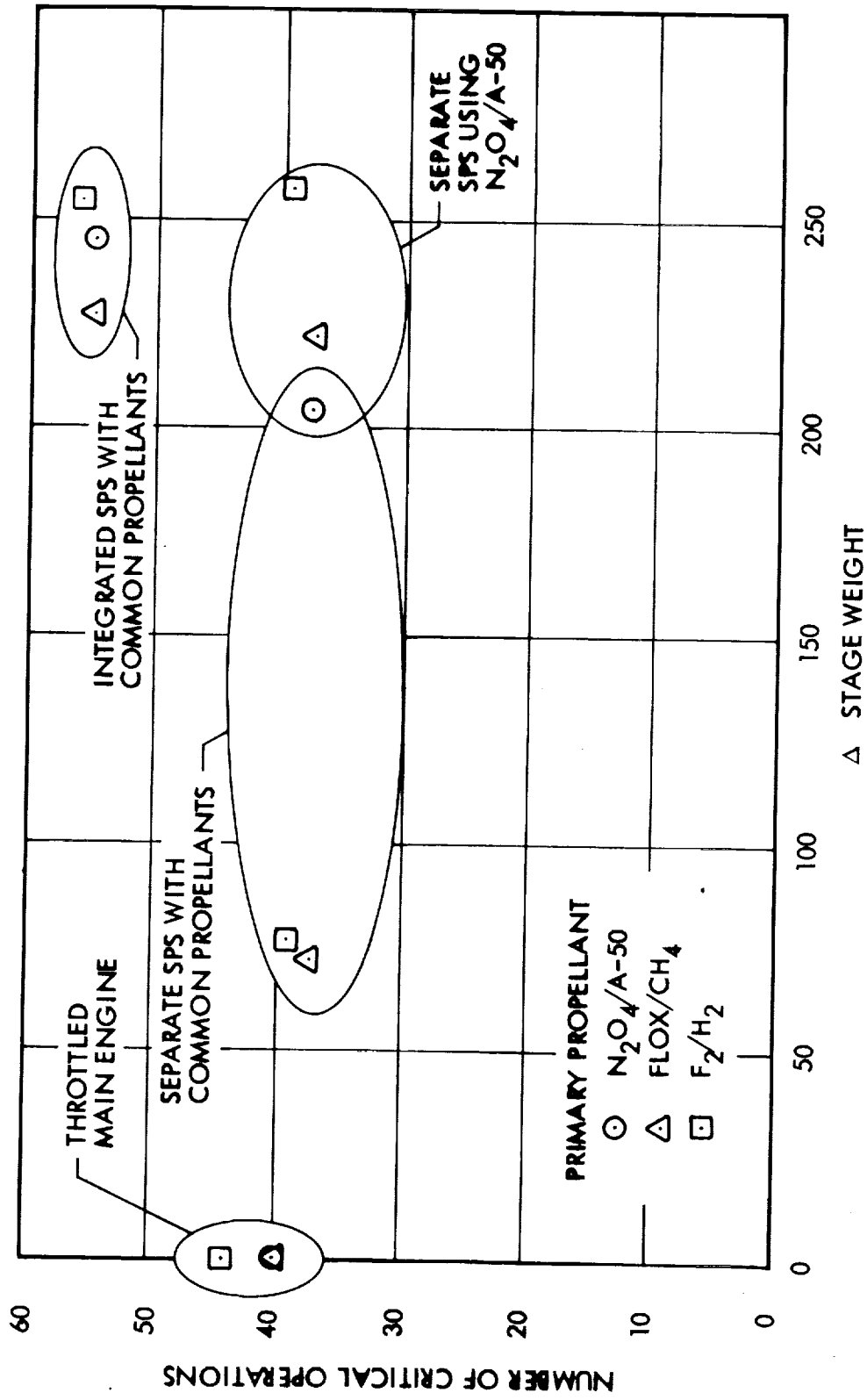


Fig. 12 Secondary Propulsion Complexity and Weight Penalty

at a small weight penalty for separate secondary systems with separate propellant tanks. The heaviest and most complex mode is the separate secondary system transferring propellant from the main tanks. N_2O_4/A-50 and FLOX/CH_4 are shown to be of equal complexity, with F_2/H_2 complexity slightly higher.

4.3 ATTITUDE CONTROL SYSTEM REQUIREMENTS DEFINITION

4.3.1 Task Objectives and Basis

The objectives of the Attitude Control System Requirements Definition task were to:

a. Define, for spacecraft and missions matched to the capability of Titan III D/Centaur, the following attitude control system requirements and effects:
   - Thruster location (moment arm)
   - Thrust level or levels
   - Duty cycle
   - Minimum impulse bit
   - Total impulse
   - Modes of operation

b. Compare, in a preliminary manner, methods of mechanizing attitude control systems to meet requirements. Consider cold gas, monopropellant, and bipropellant (separate and integrated with secondary or primary propulsion system) and estimate system weights. Recommend a concept that is attractive for analysis in greater depth.

The task was performed under the following assumptions and constraints:

- The spacecraft is sized for the launch capability of Titan III D/Centaur
- Midcourse and orbit trim corrections are accomplished by the throttled main engine
- The main engine is gimbaled for thrust vector control
- Impulse accuracy of the throttled main engine (throttled to 500 lbf) is:
  - 3σ impulse repeatability $\simeq$ 40 lb-sec
  - minimum impulse bit $\simeq$ 400 lb-sec

- Desired $\Delta V_{min} = 1 \pm 0.1$ meter/sec
- The vehicle is stabilized full time about three axes, with one axis pointed toward the sun

4.3.2 Vehicle Configurations and Inertias

Missions and spacecraft compatible with the launch capabilities of Titan III D/Centaur were assumed. These included:

Mars Orbiter – Orbit insertion stage
Jupiter Orbiter – Earth escape and orbit insertion stage
Venus Orbiter – Orbit insertion stage
Jupiter Flyby – Earth escape stage
Solor Probe – Earth escape stage
Lunar Cargo Delivery – Orbiter/Lander stage

Moments of inertia were computed about three axes for each of these configurations as a function of mission time. A summary of moments of inertia is presented in Table 20.

Figure 13 illustrates a typical spacecraft configuration at the start of interplanetary cruise and defines the reference axes used for describing mass locations. The spacecraft consists of a propulsion stage, spacecraft bus or equipment module (shown with solar panels deployed), and a lander capsule. The configuration, masses, and inertias change with mission time as propellant is consumed and the lander capsule is separated.

The requirements analysis task is based on a conventional thruster layout utilizing 12 nozzles. To avoid coupling between pitch, yaw, and roll control channels and sensitivity to spacecraft center of mass location, two thruster pairs separated by a distance L are used to generate pure control couples of ±FL ft-lb where F is the thrust in lb. In addition, the arrangement provides an attractive reliability advantage through the incorporation of independent propellant feed systems with excess capacity for each set of three pairs. Without serious loss of generality, the assumption of common F and L values for all three channels was made to make the study tractable.

Table 20

SUMMARY OF MOMENTS OF INERTIA

Item	Configuration					
	Mars 1	Mars 2	Mars 3	Jupiter 1	Jupiter 2	Jupiter 3
Weight (lb)	9700	5880	4900	6419	4199	2419
I_{xx} (slug-ft^2)	3079	1999	1753	1823	1138	696
I_{yy} (slug-ft^2)	2885	1739	1250	1188	1015	628
I_{zz} (slug-ft^2)	3580	1789	1268	1883	1333	646

Item	Configuration					
	Jup. FB 1	Jup. FB 2	Solar 1	Solar 2	Lunar 1	Lunar 2
Weight (lb)	9149	4330	6399	1580	12307	9051
I_{xx} (slug-ft^2)	2685	1264	1852	535	3890	2624
I_{yy} (slug-ft^2)	2102	677	1207	285	3980	3109
I_{zz} (slug-ft^2)	2798	677	1902	285	4119	3248

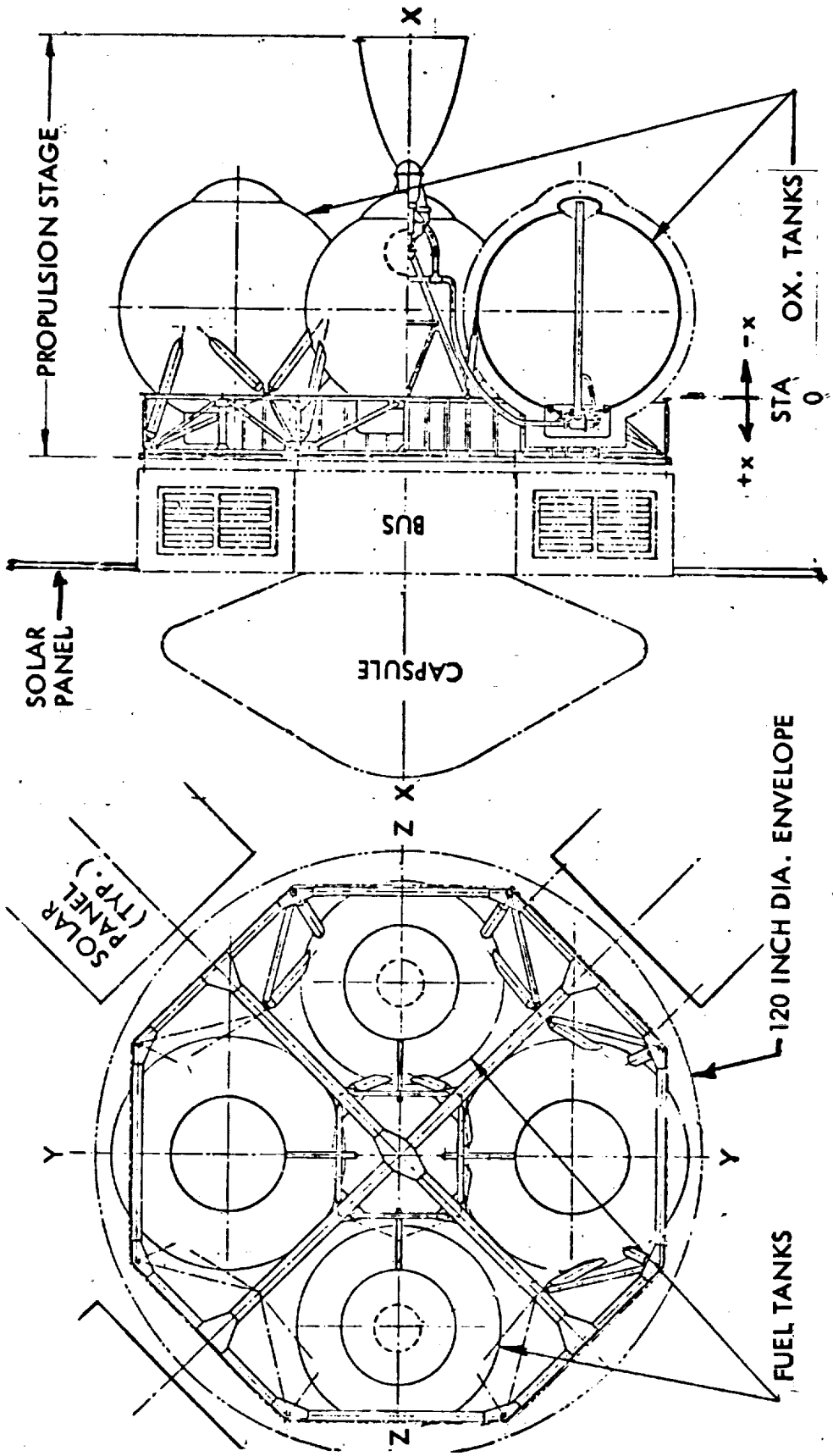


Fig. 13 Typical Spacecraft at Start of Cruise

4.3.3 Spacecraft/Controller Dynamics

Dynamic response characteristics of a spacecraft/controller combination were established by digital simulation using a computer model of one control channel and a promising controller channel. The selected controller is a sampled data version of the standard pseudo rate controller. This class of controller is particularly well adapted to implementation on a time-shared onboard digital computer, a consideration appropriate to the spacecraft generation of interest.

The simulation effort indicates that suitable controller parameter values can be found to produce satisfactory dynamic performance for all other system parameter combinations of interest. Consequently, changes in thruster force level, minimum on-time, spacecraft moment of inertia, etc., can be compensated by suitable adjustment of controller parameter values. Of equal value was the demonstration that well defined maneuvers can actually be performed with propellant expenditures that exceed the ideal value by less than 25 percent.

4.3.4 External Torque Assumptions

The primary continuous disturbing torques of interest in computing attitude control system requirements are solar torques. Precise evaluation of solar torques is both difficult and strongly dependent on spacecraft design details. In addition, torque history during cruise mode depends to some extent on spacecraft orientation history in the characteristic limit cycle motion. In view of these practical difficulties for a generalized study, the usual recourse of designing to a constant level large enough to serve as a practical bound is employed. Based on Lockheed experience the following values were estimated for earth-moon space.

$$M_x = 0.2 \times 10^{-5} \text{ ft-lb}$$

$$M_y = 1.0 \times 10^{-5} \text{ ft-lb}$$

$$M_z = 1.0 \times 10^{-5} \text{ ft-lb}$$

In the case of Mars and Jupiter missions, application of the above numbers is additionally conservative since solar pressure depends on the inverse square of distance from the sun. On the other hand, the solar probe mission terminates at a distance of 0.2 astronomical units from the sun where solar pressure is a factor of 25 higher than in earth-moon space. Accordingly, an average torque that gives the same time integral over the mission as a parabolically increasing torque (based on a conservative constant speed approach to the sun) must be 9 times the actual initial torque. Therefore, the following values were used for the solar probe mission.

$$M_x = 1.8 \times 10^{-5} \text{ ft-lb}$$
$$M_y = 9.0 \times 10^{-5} \text{ ft-lb}$$
$$M_z = 9.0 \times 10^{-5} \text{ ft-lb}$$

Solar torques may be augmented in orbital cruise modes by gravity gradient torques. Gravity gradient torques depend on the spacecraft moment of inertia tensor, together with spacecraft orientation relative to the local vertical. Since the spacecraft remains sun/Canopus oriented while cruising in orbit, the relative orientation has large magnitude, periodic variations so that many of the cyclical motions induced by these torques may be within deadbands and thus not induce control response. Nevertheless, the combined solar and gravity gradient torque envelopes were increased for orbital cruise to account for an estimated secular effect.

$$M_x = 1.0 \times 10^{-5} \text{ ft-lb}$$
$$M_y = 5.0 \times 10^{-5} \text{ ft-lb}$$
$$M_z = 5.0 \times 10^{-5} \text{ ft-lb}$$

Disturbing torques generally exist about all body axes during periods of main engine burn. A ground rule specified in paragraph 4.3.1 is that the main engine has a thrust vector control system to compensate for pitch and yaw disturbance torques. It is further assumed that the pitch and yaw attitude control thrusters are normally disabled during main engine burns to prevent channel response with attendant expenditure of

attitude control propellant. The roll channel of the attitude control system must counteract engine-generated roll disturbance torques, however. Values for these swirl torques are based on Mariner data scaled by engine thrust level, resulting in 3.0 ft-lb for a 5,000 lb thrust engine.

4.3.5 Attitude Control System Requirements and Modes of Operation

Operating Modes. The attitude control system operating modes which significantly affect propellant expenditure are: tipoff rate removal, acquisition transients and searches, commanded turns, transit cruise limit cycle, orbital cruise limit cycle, and counteracting main engine swirl torque.

ACS Propulsion Data. A review of thrusts and propellant performance capabilities was made for thruster sizes from less than 0.1 lbf to 100 lbf and for cold N_2 gas, N_2H_4, N_2O_4/MMH, FLOX/CH_4, and F_2/H_2 propellants.

It was concluded that in the thrust range of interest for limit cycle operation (less than 0.1 lbf) only the cold N_2 gas system and a cold N_2H_4 gas system were attractive. Performance of a system in this low-level thrust range could be improved by heating the gas, but this alternative was not analyzed. Specific impulse of the cold N_2 gas was assumed to be a constant 65 lbf-sec/lbm for all thrust levels and thruster on-times of interest. Specific impulse of cold N_2H_4 gas was assumed to be a constant 100 lbf-sec/lbm.

Operation of high-level thrusters was assumed to occur only in the steady state during initial orientation, reorientation, or maneuvers. Steady state specific impulse was assumed to vary as shown on Fig. 14 for the propellants considered. Specific impulse for a high-level thrust cold N_2 gas system was assumed constant at 65 lbf-sec/lbm, identical to that for the low-level system.

In the analysis of ACS propellant requirements, a cold N_2 gas low-level system was combined with each candidate high-level system for alternate combinations of mission,

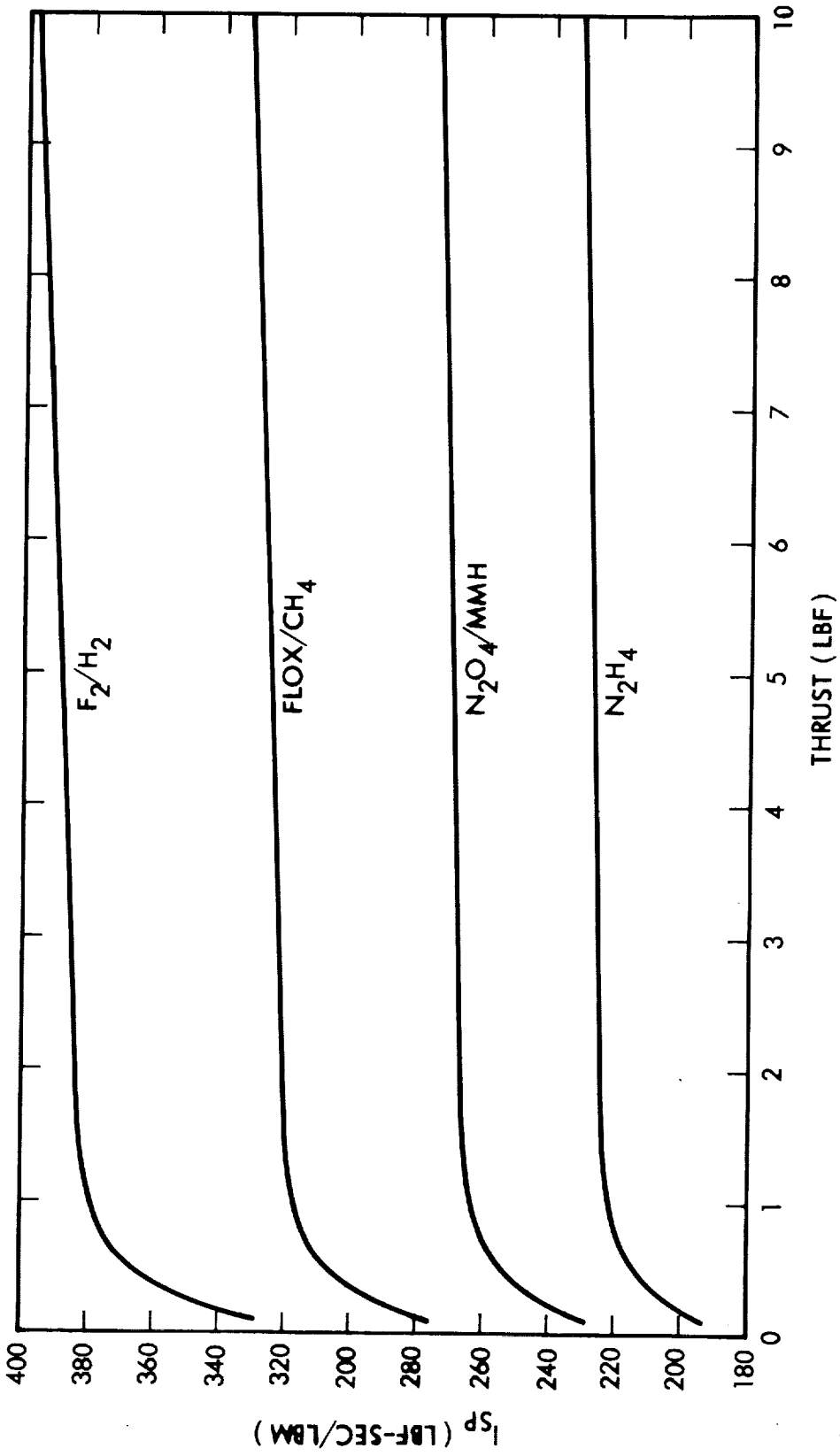


Fig. 14 Thruster Steady State Performance

thrust level, thruster moment arm, low-level thruster on-time, and limit cycle deadband. Propellant requirements are presented later in this section.

Basic Assumptions. The assumptions itemized below complete the necessary data base for the numerical study.

(1) A common thruster separation distance, L, is used for all three control channels.
(2) A common thruster force, F, is used for all three control channels except as noted in items 3 and 4.
(3) Two separate thruster force levels are employed for all missions except the intermediate size lunar lander where a single level is used.
(4) The pitch and yaw control channels are disabled during main engine burn periods while an independent thrust vector control system controls pitch and yaw attitude.
(5) The ratio of high level to low level thrust is 20 for all missions.
(6) All extended duration cruise modes are performed with low-level thrusters using cold nitrogen gas with a specific impulse of 65 sec.
(7) All non-cruise modes are performed with high-level thrusters.
(8) The maximum attitude control system torque capability of interest is 20 ft-lb.
(9) Computer outputs do not include an allowance for leakage or any reserve to handle malfunctions.
(10) Three representative thruster separation distances of interest are L = 5, 10, 25 ft.
(11) Three representative minimum on-times of interest (applies only to low-force-level nitrogen thrusters) are Δ = 0.025, 0.050, 0.075 sec.
(12) Three cruise mode deadband levels of interest are D = ±0.5, 1.0, 5.0 deg. An exception is the intermediate size lunar lander spacecraft where one half of these values are used for orbital cruise.

Requirements Analysis Program. A Univac 1108 program in Fortran V was modified to perform the requirements calculations of each mission and to prepare SC-4020 plotter

tapes for graphical solution display. The purpose of the program was to generate graphic displays of mission control requirements that show the effects of variations in the following four parameters:

(1) Thruster force level (both high and low levels constrained by a fixed 20:1 ratio)
(2) Thruster minimum on-time, (0.25, 0.50, and 0.75 sec)
(3) Thruster separation distance, (5, 10, and 25 ft)

Based on the selected plot layout, the first two parameters above were automatically varied over a specified range during a single computer run, while fixed values were input for the last two. Accordingly, each plot corresponds to a specific combination of deadband and thruster separation distance; a set of several computer runs is required to display the effect of variation in these parameters.

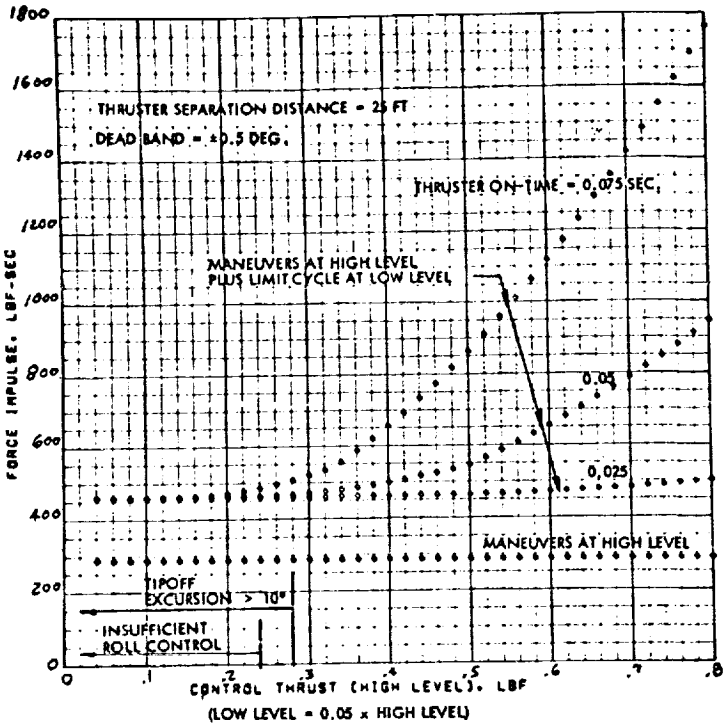
A sample set of computer outputs is presented in Fig. 15 for a thruster separation of 25 ft, limit cycle dead bands of ±0.5, ±1.0, and ±5.0 deg, thruster on-times of 0.25, 0.50, and 0.75 sec, varying thrust levels, and alternate propellants.

4.3.6 ACS Requirements Summary

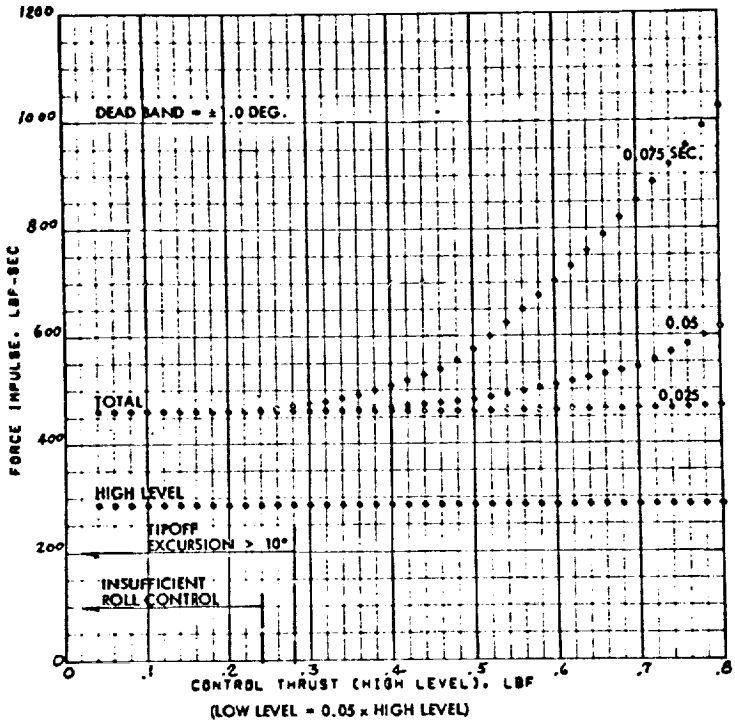
From an analysis of the attitude control systems requirements calculations and computer outputs, the following general observations and conclusions are made:

1. Two levels of thrust are required, for most of the missions analyzed, in order to keep ACS propellant requirements to an acceptable low level. For the lunar lander mission the high-level thrusters could be used for the short transit and orbit times, as well as for maneuvers, at a relatively small penalty. High-level thrusters in the 1 to 5 lb thrust range are suitable for all the missions, while low-level thrust in the 0.05 lbf range is recommended (0.3 lbf for the lunar lander).
2. Limit cycle operation should be accomplished with the smallest combination of thrust and thruster on-time that will overcome external torques. Specifically, assuming a thruster separation of 10 ft, minimum inputs bits (MIB) of

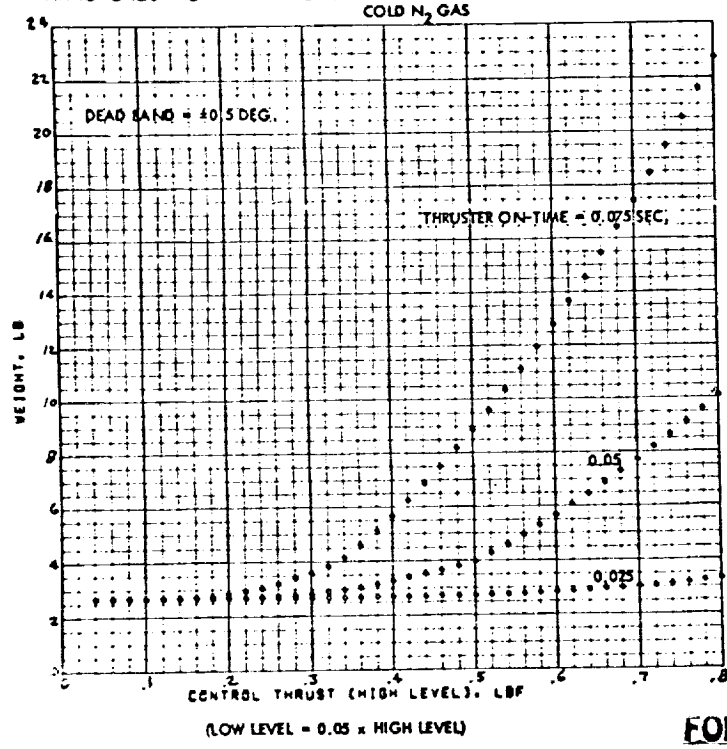
MARS ORB. 3 TOTAL CONTROL IMPULSE


THRUSTER SEPARATION DISTANCE = 25 FT
DEAD BAND = ±0.5 DEG.
THRUSTER ON-TIME = 0.075 SEC.
MANEUVERS AT HIGH LEVEL PLUS LIMIT CYCLE AT LOW LEVEL
0.05
0.025
MANEUVERS AT HIGH LEVEL
TIPOFF EXCURSION > 10°
INSUFFICIENT ROLL CONTROL
FORCE IMPULSE, LBF-SEC
CONTROL THRUST (HIGH LEVEL), LBF
(LOW LEVEL = 0.05 x HIGH LEVEL)

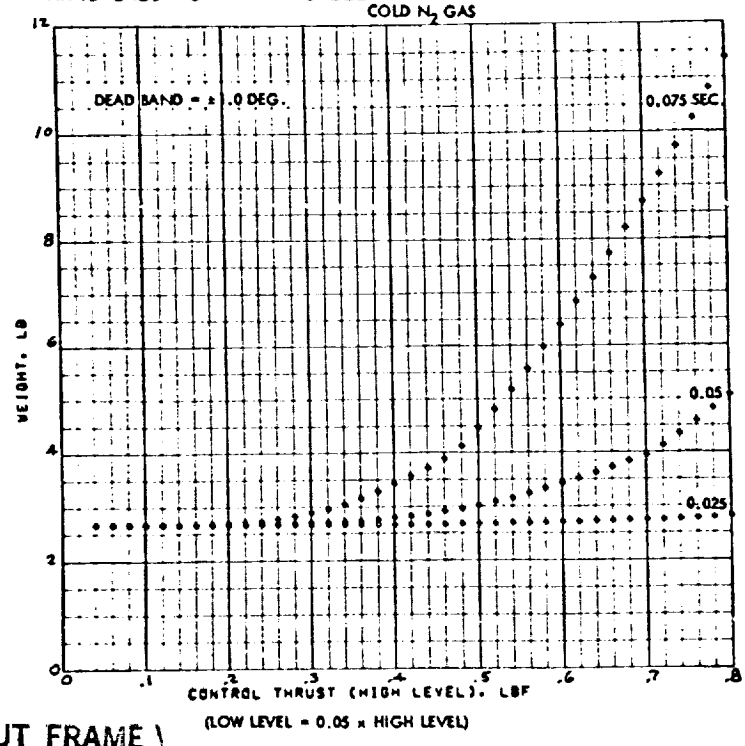
MARS ORB. 6 TOTAL CONTROL IMPULSE



MARS ORB. 3 PROPELLANT WEIGHT (LOW LEVEL) COLD N_2 GAS



MARS ORB. 6 PROPELLANT WEIGHT (LOW LEVEL) COLD N_2 GAS



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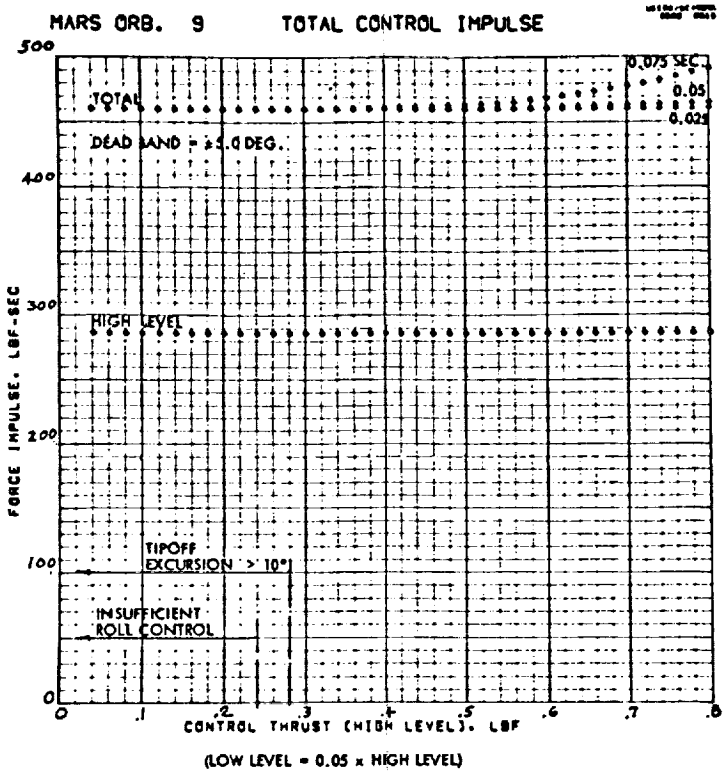

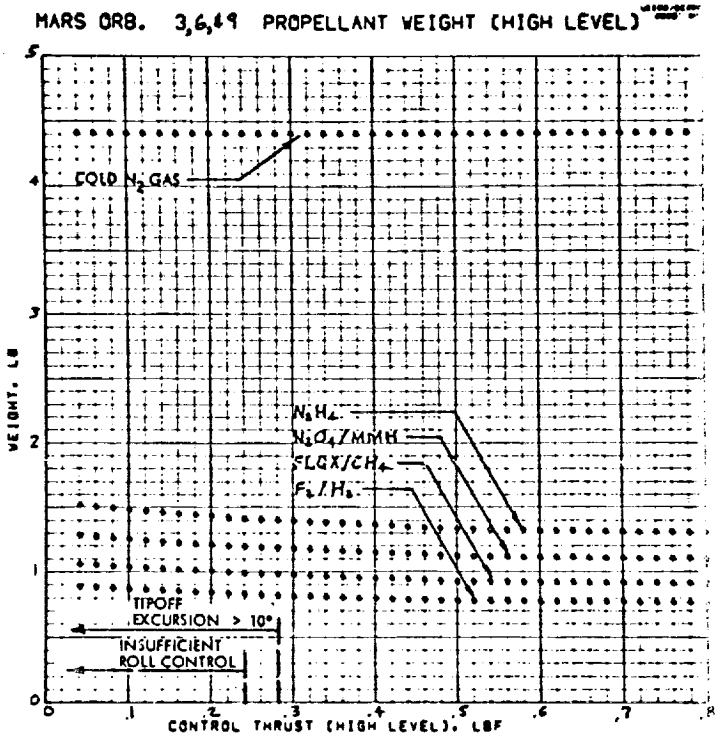

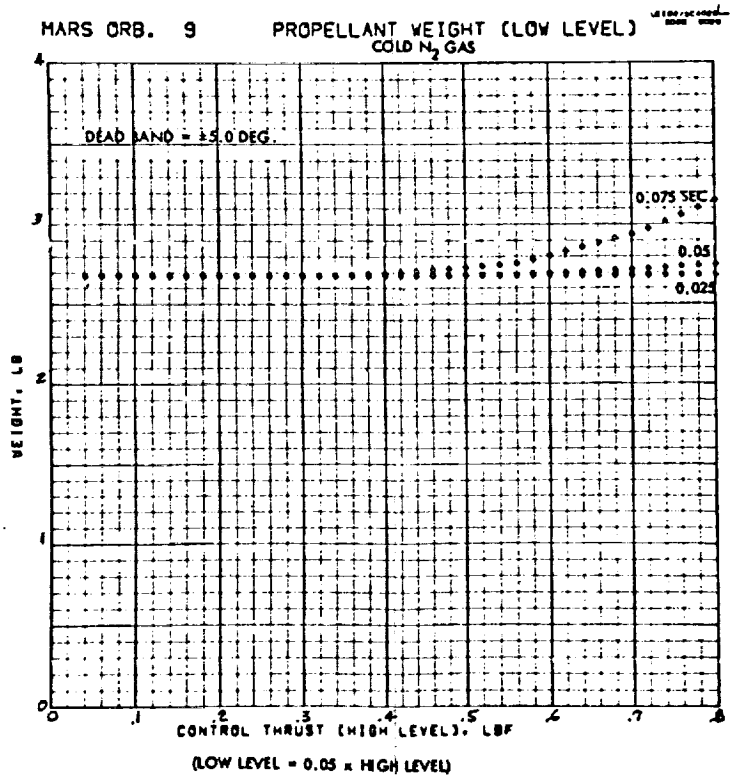

Fig. 15 Mars Orbiter ACS Requirements
Thruster Separation = 25 ft.

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0.001 lbf-sec or less were found desirable for the Solar Probe mission, 0.00125 lbs-sec for the Jupiter Orbiter and Jupiter Flyby, 0.0015 lbf-sec for the Mars Orbiter, and 0.0075 for the Lunar Lander.

3. Thruster location should be as far as practical from the center of mass and rotation. For example, if solar panels are installed then location of thrusters at panel extremities should be considered. Note, though, that there is an optimum Minimum Impulse Bit value associated with each thruster location, all other mission and spacecraft conditions being fixed. If that optimum MIB cannot be achieved, then some shorter moment arm can be found which will be more economical of ACS propellant.
4. Thruster on-time should be as low as practical for limit cycle operation. An on-time of 0.025 sec was chosen as the lower practical limit for the analysis, although it is known that somewhat lower values are achievable for specific propellants and systems.
5. Limit cycle deadband should be as large as mission requirements allow, especially during long-term cruise or orbit.
6. High-level propellant requirements are mildly sensitive to thrust level and directly proportional to specific impulse.
7. Mars Orbiter high-level propellant requirements vary from 21 percent (for F_2/H_2) to 60 percent (for cold N_2 gas) of total ACS propellant requirements when combined with an optimized low-level system using cold N_2 gas.

A summary of normal or recommended ACS parameters is presented in Table 21 together with estimated weights for propellants and total ACS systems. For all missions the use of N_2H_4 monopropellant for the high-level system resulted in the lightest total weight. The estimation of system weights is described in the following subsection.

4.3.7 ACS System Weight Estimates

A preliminary examination of methods for mechanising ACS systems was made in order to compare system weights and recommend a concept.

Table 21

SUMMARY OF NOMINAL OR RECOMMENDED ACS PARAMETERS AND ESTIMATED WEIGHTS

Mission	Lunar Lander	Solar Probe	Jupiter Orbiter	Jupiter FlyBy	Mars Orbiter
Nominal Limit Cycle Deadband (± deg)	0.5	0.5	1.0	1.0	1.0
† Nominal Thruster Separation Distance Assumed (ft)	10	10	10	10	10
Recommended Limit Cycle Pulse Width (sec)	≤.025	≤.025	≤.025	≤.025	≤.025
Recommended Low-Level Thrust (lbf)	≤.30	≤.06	≤.04	≤.05	≤.05
Recommended High-Level Thrust (lbf)	1 to 5	1 to 5	1 to 5	1 to 5	1 to 5
Low Level Impulse Required (lb-sec)	21	454	2503	801	370
High Level Impulse Required at 1.4 lbf (lb-sec)	1972	717	460	508	564
Sum of Low and High-Level Impulse (lb-sec)	1993	1171	2963	1309	934
*Low Level Propellant (N_2 gas) (lb)	0.32	6.99	38.51	12.32	5.70
*High-Level Propellant (lb)					
N_2 gas	30.35	11.03	7.08	7.81	8.68
N_2H_4	8.83	3.21	2.06	2.30	2.55
N_2O_4/MMH	7.46	2.71	1.74	1.94	2.15
FLOX/CH_4	6.18	2.25	1.44	1.61	1.79
F_2/H_2	5.16	1.88	1.20	1.34	1.49
Estimated Total ACS System Wt (lb)					
Low-Level/High-Level (1.4 lbf)					
N_2 Gas/N_2 Gas	97.4	68.8	155.9	73.3	62.9
N_2 Gas/N_2H_4	48.6	55.1	125.9	66.6	54.5
N_2 Gas/N_2O_4/MMH	59.5	72.9	145.0	84.5	72.4
N_2 Gas/FLOX/CH_4	63.2	72.4	144.7	84.2	72.0
N_2 Gas/F_2/H_2	62.4	73.2	145.6	86.1	72.7
N_2 H_4 Gas/N_2H_4	48.5	69.4	50.0	44.0	47.0

* Impulse Propellant With No Redundancy

† Thruster Separation of 20 to 25 ft, easily achievable with spacecraft using solar panels for power, would reduce thrust levels and impulse requirements in direct proportion to the increased separation distance.

Cold N_2 Gas and Cold N_2H_4 Gas.

The only systems examined that were capable of delivering the very low thrust and minimum imput bit required were cold N_2 gas and cold N_2H_4 gas.

The cold N_2 gas system is the most widely used to date, is inherently simple, and easily storable. Cycle life is limited by pressure regulators and solenoid valves and contamination is one of the most severe problems, leading to leakage. The dominant environmental problem is vibration with the pressure regulator most susceptible to vibration failure. Attractive applications are in thrust levels of one lb or less. A cold N_2 gas system, with two levels of thrust controlled by varying regulated pressure, can meet all requirements for the missions and spacecraft studied.

Use of cold N_2H_4 gas for low-level thrust in combination with N_2H_4 monopropellant high-level thrusters was examined briefly. This system has the dual advantages of use of a common propellant coupled with the lowest estimated weight for each mission. For the most demanding mission (Solar Probe) the ACS system weight is estimated at 69.4 pounds as compared to 125.9 pounds for the next lightest system which uses cold N_2 gas for low-level and N_2H_4 for high level control.

Heating of the cold gases was not evaluated, but may prove attractive in increasing the delivered specific impulse.

N_2H_4 Monopropellant

N_2H_4 monopropellant is useful and attractive in the 1 to 5 lb thrust range and is capable of delivering a specific impulse of about 235 sec steady state. The main problem is keeping the propellant from freezing (at +34° F). This system is not suitable for the low-level ACS requirements studied but yields the lightest weights for the high-level thrust applications, as shown by Table 21.

Bipropellants

Bipropellants are most useful at thrust levels exceeding 5 or 10 lb. It is therefore marginal to consider bipropellants even for high-level ACS thrusters for the configurations studied. I_{sp} values varying from about 275 sec for MMH to about 400 sec for F_2/H_2 are realizable in the 5 to 10 lb thrust range. Problems with low thrust bipropellant engines includes hard-start, valve leakage in hard vacuum, injector heat soakback in buried installations, and orifice plugging.

Total ACS system estimated weights are shown in Table 21 for alternate missions and propellant combinations. Assumptions made in preparing these estimates are:

1. **10 percent excess propellants**
2. **Thruster redundancy on both low and high thrust levels. Use 4 squib valves to transfer**
3. **Titanium tanks used for all gases and liquids except aluminum tanks used for LH_2 and LF_2**
4. **1.5 safety factor used in all tanks. 10 percent ullage in all liquid tanks**
5. **Minimum tank weight of 0.3 lb**
6. **N_2 thrusters have 15 psia chamber pressure. Monopropellant and bipropellant thrusters operate at 300 psia (except F_2/H_2 100 psia)**
7. **H_2 and F_2 are gaseous when introduced into chamber. Chamber pressure is 100 psia. Two pressure regulators required for separate gases**
8. **No instrumentation weight included**

Note from Table 21 that the alternate bipropellants have essentially constant system weight for a given application in spite of differences in I_{sp}.

4.3.8 Recommendations

It is recommended that further examination of ACS systems for the spacecraft and missions studied concentrate on the following alternatives:

1. Cold N_2 gas for both low and high-level thrust as the most simple system

2. Cold N_2 gas for low-level and N_2H_4 for high-level thrust as a system within current technology and of lighter weight than all cold N_2 gas
3. Cold N_2H_4 gas for low-level and N_2H_4 for high-level thrust as the lightest weight system
4. Consideration of heating the cold gases to improve I_{sp} and reduce system weight

4.4 GROUND OPERATIONAL REQUIREMENTS AND PROBLEMS

4.4.1 Task Objectives and Basis

The objectives of the ground operational requirements analysis task were to evaluate and compare operations requirements, facilities requirements, and complexity for a mission of current interest and for several alternate propellants, and to compare propellant loading on the launch pad with loading in a remote explosive-safe facility. Ground rules and assumptions specified for the analysis included:

- Compare alternate propellant combinations including F_2/H_2, O_2/H_2, FLOX/CH_4, OF_2/B_2H_6, F_2/NH_3, and N_2O_4/A-50.
- Assume that the propulsion stage is designed as a modular part of a Mars Orbiter spacecraft system to be launched by Titan III D/Centaur from complex 40-41 at Cape Kennedy.

Figure 16 shows a plan view of the Integrate Transfer Launch area at the Eastern Test Range (Cape Kennedy). These facilities were designed and built with the operational objective of accomplishing assembly and checkout of the complete vehicle, including spacecraft, in a central area. This central area, identified as the Vertical Integration Building (VIB), is shown in the figure. The functions performed in the VIB are:

- Assembly and checkout of the Titan core vehicle
- Assembly and checkout of the Centaur vehicle
- Assembly and checkout of the spacecraft and payloads vehicle
- Integrated mission simulation test

ITL LAUNCH COMPLEXES 40 AND 41

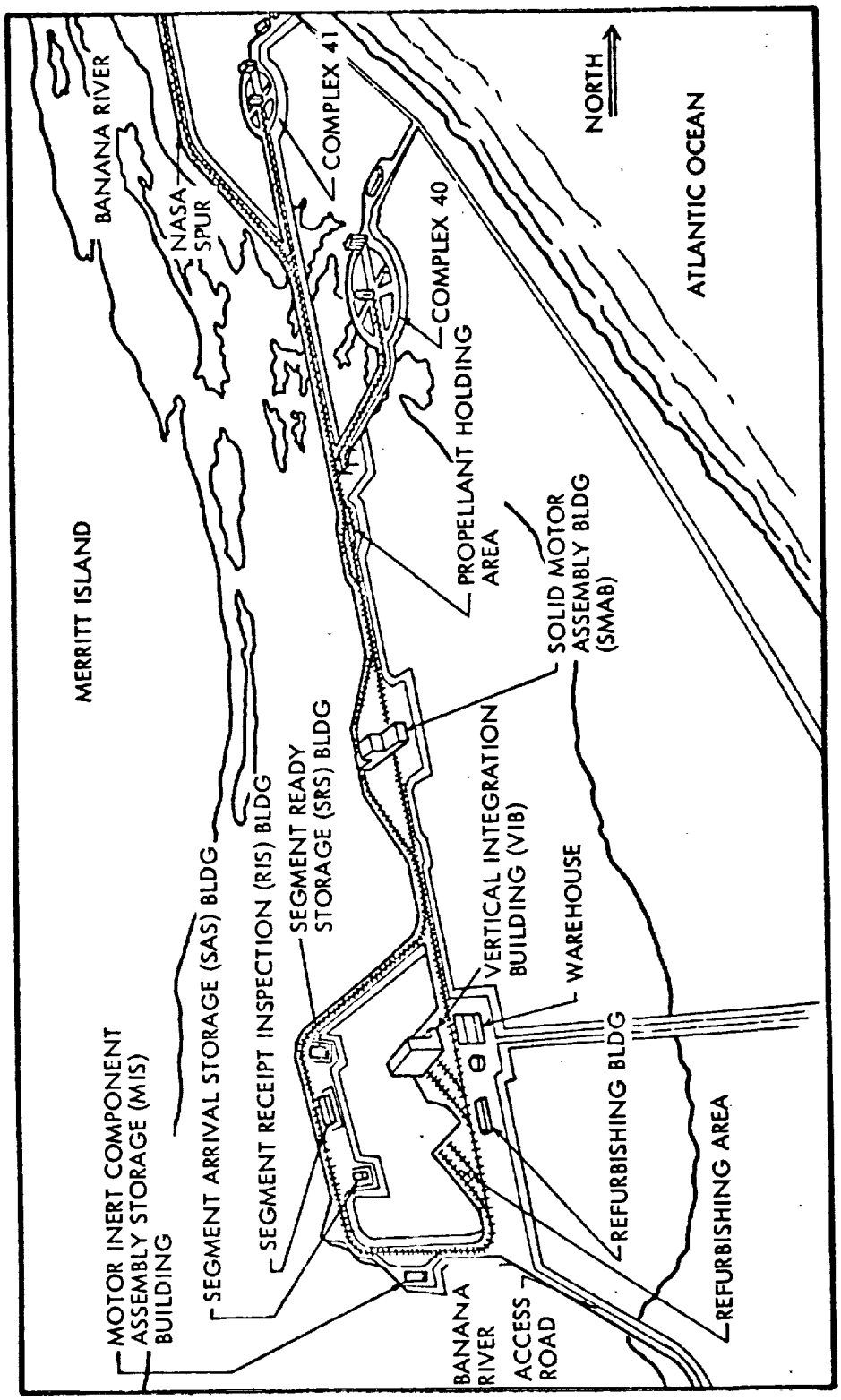


Fig. 16 Integrate Transfer Launch Area, Plan View

Upon completion of integrated mission simulated tests the Titan core/Centaur/spacecraft vehicle is transported to the Solid Motor Assembly Building (SMAB) where the Titan solid motors are mated to the core vehicle. Finally, the complete, integrated vehicle is transported to the launch pad (40 or 41) for final pre-launch servicing, propellant loading, countdown, and launch.

4.4.2 Ground Operations

An analysis was made of the operational sequences required for propellant handling from initial tanking to final vehicle topping. This included studying problems associated with ground transport, storage at the launch complex, vehicle tank conditioning, transfer from storage to vehicle, thermal conditioning, insulation purging, and toxicity and hazard. Specific steps required for each of these items and each propellant were defined. An assessment was made of the requirements for a ground hold, and the recycle time and steps needed for a complete turnaround. Penalties associated with a no-vent condition on the pad were evaluated and a comparison made of propellant loading at the pad vs loading in a remote explosive-safe facility.

4.4.2.1 On-Pad Vs Off-Pad Propellant Loading. In an operational situation many advantages accrue to a spacecraft propulsion system that can be loaded, checked out, and buttoned up prior to moving the vehicle to the launch pad. This mode of operation has been used quite successfully for such spacecraft as Surveyor and Lunar Orbiter using earth-storable propellants. With the introduction of space-storable or cryogenic propellants in the spacecraft propulsion system, this approach is complicated by the need for thermal control to maintain the propellants in the liquid state. The advantages of off-pad propellant loading include:

- Launch pad occupancy time for spacecraft functional checks, leak checks, propellant tank preconditioning, etc., is considerably reduced.
- The possibility of delays or aborts on the launch pad being caused by propulsion system problems are minimized since final verification of readiness is with propellants loaded.

- Loading and checkout at a remote site is much more convenient and can be accomplished on a more leisurely schedule well before launch time.
- Backup spacecraft can be held in a completely loaded and checked-out state, ready to be transported to the pad in exchange for a malfunctioning vehicle.
- Provision of a thermal conditioning unit can eliminate the need for propellant venting, vent gas disposal, or propellant tank topping on the pad.
- The hazards of transferring spacecraft propellants at the pad are eliminated.

The disadvantages of off-pad propellant loading include:

- Thermal conditioning units must be added for all space-storable and cryogenic propellants.
- Spacecraft flight weight will probably increase slightly to accommodate the flight portion of the thermal conditioning system.
- Handling of the fully loaded spacecraft from remote site to launch vehicle mating will be more difficult due to the increased weight and to the greater hazards introduced.

The final decision to load propellants on-pad or off-pad will require a more complete definition of the vehicle system. It appears that off-pad propellant loading for spacecraft of the Mars Orbiter size, using any class of propellant, is an attractive goal. The key problem lies with the design of the propellant thermal control system. An analysis of vented and non-vented systems was made and the resulting heat inputs, boiloff rates, cooling flowrates, and flight hardware weight penalties are shown in Table 22.

4.4.2.2 Propellant Hazards Comparison. Each of the propellants under study presents some hazard to personnel. The least hazardous are H_2, O_2, and CH_4 which are all non-toxic and require no respiratory protection, but do require body protection to prevent frost type burns. All of the remaining propellants are quite toxic with threshold limits varying from 500 ppm for NH_2 to 0.05 ppm for OF_2, based on values recommended by the American Conference of Government Industrial Hygienists, 1968. Special

Table 22

GROUND HOLD ANALYSIS RESULTS

		VENTED SYSTEM	NON-VENTED SYSTEM		
PROPELLANT	GROUND ENVIRONMENT HEAT INPUT (BTU/HR)	BOILOFF RATE, (LB/HR)	COOLING METHOD	COOLING FLOWRATE (LB/HR)	ESTIMATED FLIGHT HARDWARE INCREASE (LB)
F_2	6,100	82.4	LIQUID N_2 SPRAY COOLING	72.5	2.0
H_2	16,300	84.0	*COLD HELIUM	685.0	6.0
O_2	9,220	100.4	LIQUID N_2 SPRAY COOLING	109.5	3.1
H_2	25,590	131.8	*COLD HELIUM	1070.0	6.0
FLOX	6,364	85.4	LIQUID N_2 SPRAY COOLING	75.6	2.0
CH_4	4,302	17.4		51.2	1.6
OF_2	5,232	58.6		62.1	2.0
CH_4	4,460	18.0		53.0	1.6
F_2	5,984	80.6		71.1	1.9
NH_3	1,212	2.1		14.4	1.5
OF_2	5,538	62.0		65.8	2.0
B_2H_6	3,456	15.6		41.1	1.7
N_2O_4	NEGLIGIBLE	NEGLIGIBLE	NONE	NONE	NONE
A-50	NEGLIGIBLE	NEGLIGIBLE	NONE	NONE	NONE

*CLOSED CYCLE HELIUM HEAT EXCHANGER WITH NO REFRIGERANT LOSS

breathing equipment is required. Toxicity of F_2 and FLOX is stated as 0.1 ppm, though recent studies indicate this may be raised to 1.0 ppm. All will damage body tissue through burns or dermatitis, and body protection is required.

A summary of toxicity and body tissue personnel hazards is presented in Table 23.

All of the propellants are potentially hazardous through fire, explosion, and stability characteristics. Each of the fuels will burn in the presence of air. B_2H_6 is the most easily ignited in air, with flammability limits between 0.9 and 93 percent by volume and auto ignition temperature of 300°F. This is followed closely by A-50 at 2.0 to 90 percent and 450°F. The least hazardous is CH_4 at 4.0 to 15 percent and 1200°F.

All of the fluorinated oxidizers are highly reactive with most substances at room temperature. All are stable. Table 24 summarizes fire, explosion, and stability characteristics.

4.4.2.3 Propellant/Materials Compatibility. Suitable materials are available for storage tanks, transfer lines, valves, seals, etc., for use with each of the candidate propellants. Passivation of all equipment is required for most of the propellants. Table 25 summarizes propellant/materials compatibility for each oxidizer and fuel for representative metals and non-metals. The compatibility of aluminum with B_2H_6 has not been definitely established, but preliminary results of a current investigation at Stanford Research Institute under JPL contract 951584 are favorable.

4.4.3 Operations and Facilities Complexity

A summary of the factors that affect the ground operational requirements was made and the relative complexity of operational tasks and facility elements for alternate systems evaluated. The total number of ground operational tasks and the total number of facility elements are presented in Table 26 for each propellant combination as one measure of complexity.

Table 23

PERSONNEL HAZARDS COMPARISON

PROPELLANT	*TOXICITY (THRESHOLD LIMIT VALUE)	RESPIRATORY PROTECTION REQUIRED	EXPOSURE TO BODY TISSUE	BODY PROTECTION REQUIRED
B_2H_6	0.1 ppm/0.1 mg/m^3	Self cont. breathing equipment	Skin dermatitis	Gloves and face
H_2	None	None	Frost type burns	Gloves and face
CH_4	None	None	Frost type burns	Gloves and face
NH_3	50 ppm/35 mg/m^3	Canister mask or self-contained breathing equipment	Chemical burns	Gloves and face
A-50	1 ppm	Self contained breathing equipment	Chemical burns	Gloves and face
FLOX	0.1 ppm/0.2 mg/m^3	Self contained breathing equipment	Severe acid type burns	Full acid type suits
F_2	0.1 ppm/0.2 mg/m^3	Self contained breathing equipment	Severe acid type burns	Full acid type suits
OF_2	0.05 ppm/0.1 mg/m^3	Self contained breathing equipment	Acid type burns	Full acid type suits
O_2	None	None	Frost type burns	Gloves and face protector
N_2O_4	5.0 ppm/9 mg/m^3	Self contained breathing equipment	Acid type burns	Acid resistant gloves, boots and face

* Based on recommended values adopted by American Conference of Governmental Industrial Hygienists, 1968

Table 24

FIRE, EXPLOSION, AND STABILITY HAZARDS COMPARISON

PROPELLANT	FLAMMABILITY LIMITS IN AIR % BY VOL	AUTO IGNITION TEMP. °F (IN AIR)	GENERAL COMMENTS
B_2H_6	0.9 to 93	300°F	Very low auto ign. temp. Contaminants can cause fires at room temp. Decomposes above -112°F.
H_2	4.0 to 74.2	1075	Fire potential if spills or leaks occur – (stable)
CH_4	4.0 to 15	1200	Fire potential if spills or leaks occur – (stable)
NH_3	16.1 to 26.8	1200	Fire potential if spills or leaks occur. Disassociates above 900°F.
A50	2.0 to 90	450	Fire potential if spills or leaks occur. Decomposes above 500°F.
FLOX	———	———	High rate of reaction with fuels, water. Reacts with most substances at room temp. (stable)
F_2	———	———	High rate of reaction with fuels, water. Reacts vigorously with most substances at room temp. (stable)
OF_2	———	———	Reacts with fuels. Reacts with most substances but to lesser degree than F_2. (stable)
O_2	———	———	High rate reaction with hydrocarbons (stable)
N_2O_4	———	———	Supports combustion in contact with fuels (stable)

Table 25

PROPELLANT/MATERIALS COMPATIBILITY

		METALS							NON-METALS				
		TITAN-IUM	MILD STEEL	STAIN-LESS STEEL	ALUMI-NUM	NICKEL	COPPER	BRASS	POLY-ETHYLENE	ASBESTOS	TEFLON	KEL-F	MYLAR
OXIDIZER	FLOX	NO	NO	304L	YES	YES	YES	YES	NO	NO	YES	YES	NO
	F_2	↓	↓	304L	↓	↓	↓	YES	↓	↓	↓	↓	↓
	OF_2	↓	↓	YES	↓	↓	↓	YES	↓	↓	↓	↓	↓
	O_2	NO	NO	YES	↓	↓	YES	NO	NO	↓	↓	↓	↓
	N_2O_4	YES	YES	YES	YES	YES	NO	NO	YES	NO	YES	YES	NO
FUEL	B_2H_6	*	NO	YES	*	YES	NO	YES	YES	YES	YES	YES	YES
	H_2	YES	↓	↓	YES	YES	YES	YES	NO	NO	↓	↓	YES
	CH_4	*	↓	↓	YES	YES	↓	YES	YES	YES	↓	↓	YES
	NH_3	YES	↓	↓	NO	NO	↓	NO	YES	YES	↓	↓	NO
	A-50	YES	NO	YES	YES	NO	YES	NO	YES	NO	YES	YES	YES

*INSUFFICIENT DATA AVAILABLE

Table 26

COMPLEXITY COMPARISON

Propellant	N_2O_4/A-50	O_2/H_2	F_2/NH_3	OF_2/B_2H_6	F_2/H_2	FLOX/CH_4
Total No. of Operational Tasks	21	30	32	36	36	38
Total No. of Facility Elements	10	16	18	23	22	28

Notes:

1. FLOX mixed at the pad
2. Vehicle tanks loaded at the pad from pad propellant storage facilities

4.4.4 Task Conclusions

As a result of the ground operations study it was concluded that there should be no serious problems in handling any of the candidate propellants at the Eastern Test Range. It was further concluded that:

- Complex 40-41 at ETR can be modified for operations utilizing any propellant combination studied.
- FLOX/CH_4 is most complex and N_2O_4/A-50 is least complex for overall operations and facilities.
- No-vent ground hold systems are feasible for all propellants studied and result in negligible increases in vehicle flight weights.

4.5 PROPULSION STAGE COMMONALITY

4.5.1 Task Objectives and Ground Rules

The purpose of the propulsion stage commonality task was to: (1) investigate the feasibility of utilizing a common stage, with minimum modification, for alternate space missions and (2) to assess the sensitivity of commonality objectives to propellant combination selected. In the analysis the basic propulsion stage structure, propellant tanks, and engine systems were designed for a Mars orbiter mission. For other missions the basic stage remained essentially fixed, with insulation, surface coatings, pressurization system and meteoroid protection varied to suit the mission requirements. Propellant tank sizes were fixed by the Mars orbiter mission and propellant loading was varied for other missions if this improved performance.

The baseline propulsion stage was designed for orbit injection at Mars of a Mars orbiter sized for interplanetary injection by a Titan IIID/Centaur launch vehicle. The mission description and propulsive maneuvers required are as follows:

- 1973 Mars Orbiter/Lander
- 205-day duration with 195-day interplanetary trip, and orbit trim after 10 days in orbit about Mars
- 6,950 ft/sec. total velocity required of the stage
- Parking orbit ascent mode to 100 nm with up to 90 min. in earth orbit

Four propulsive steps are:

- First midcourse = 50 ft/sec @ T = 3 days
- Second midcourse = 17 ft/sec @ T = 165 days
- Orbit insertion = 6555 ft/sec @ T = 195 days
- Orbit trim = 328 ft/sec @ T = 205 days

Three propellant combinations were considered in detail, including one cryogen (F_2/H_2, pump fed), one space-storable (FLOX/CH_4, pump fed), and one Earth-storable (N_2O_4/A-50, pressure fed). In addition one pressure fed space storable (OF_2/B_2H_6) stage was examined to a lesser extent. Three engine companies – Aerojet-General, Pratt and Whitney Aircraft, and Rocketdyne provided technical data for the propellants and engine systems used in this task.

A propulsion module was first defined and its propulsion requirements determined for the Mars Orbiter mission. This was accomplished by utilizing the Mars Orbiter spacecraft and optimized pump-fed propulsion stage configurations issuing from Phase I of this study and scaling them to match the capability of a Titan IIID/Centaur launch vehicle to perform the nominal Mars Orbiter mission. The stage was also resized to adapt to the 10-foot diameter Centaur. Payload weight and dimensions were scaled down accordingly, but no attempt was made to reconfigure the spacecraft or to analyze spacecraft details such as capsule payload requirements vs aerodynamic drag parameter.

An analysis was then made to determine the feasibility of using the commonality stage to perform the following missions when launched from Earth by the Titan IIID/Centaur:

(a) Mars Orbiter-Orbit Injection Stage (Baseline)
(b) Venus Orbiter-Orbit Injection Stage
(c) Lunar Cargo Delivery-Orbiter/Lander Stage
(d) Jupiter Flyby-Earth Escape Stage
(e) Solar Probe to 0.20 AU-Earth Escape Stage
(f) Mars Orbiter – stage sized to Titan IIID/Centaur but used on Titan IIID with ascent burn
(g) Mars Orbiter – ascent burn and orbit inject stage optimized for Titan IIID/Centaur
(h) Jupiter Orbiter – orbit injection stage used first at Earth in ascent burn mode

The characteristics of the selected missions were defined and payload capabilities of the commonality stage in performing these missions was estimated.

Having established that the commonality stage payloads were potentially very attractive, the commonality stage design was initiated. The basic structural, propulsion, and thermodynamic characteristics were defined in detail and an optimization procedure was carried out to completely describe the design parameters and performance characteristics of the Mars orbiter vehicle. This module was then flown on the other selected missions and a new optimization computed allowing the insulation thickness, pressurization requirements, and ullage volume to vary. Preferential spacecraft orientation and shadow shielding were used where advantageous. Modifications required to the propulsion stage structure, insulation, propellant loadings, etc., were identified and mission performance recalculated to determine a refined payload capability for each stage and mission combination. Concurrently, optimum sizing for stages used in the ascent-burn mode at Earth departure prior to use for midcourse corrections and for orbit insertion at Mars were determined. Payload capability in Mars orbit for a stage sized to Titan IIID/Centaur, but launched by Titan IIID and used in the ascent burn mode, was also determined.

Finally, the commonality characteristics were evaluated and the commonality concept analyzed. This was accomplished by making a detailed comparison of the relative performance, operational, and complexity/simplicity advantages of each propellant combination, in comparison with the remaining propellants, for each mission.

4.5.2 Initial Stage Sizing

The initial stage sizing was accomplished by optimizing the Mars orbiter vehicle for each propellant combination and within the Titan IIID/Centaur booster capability shown in Fig. 17. This figure indicates that a payload of 9700 pounds can be injected to the Mars transfer velocity of 38,540 feet per second. This 9700 pounds encompasses the payload, spacecraft, and propulsion module. Only the propulsion module design was analyzed in detail while the spacecraft, science, capsule, etc. were lumped as useful payload.

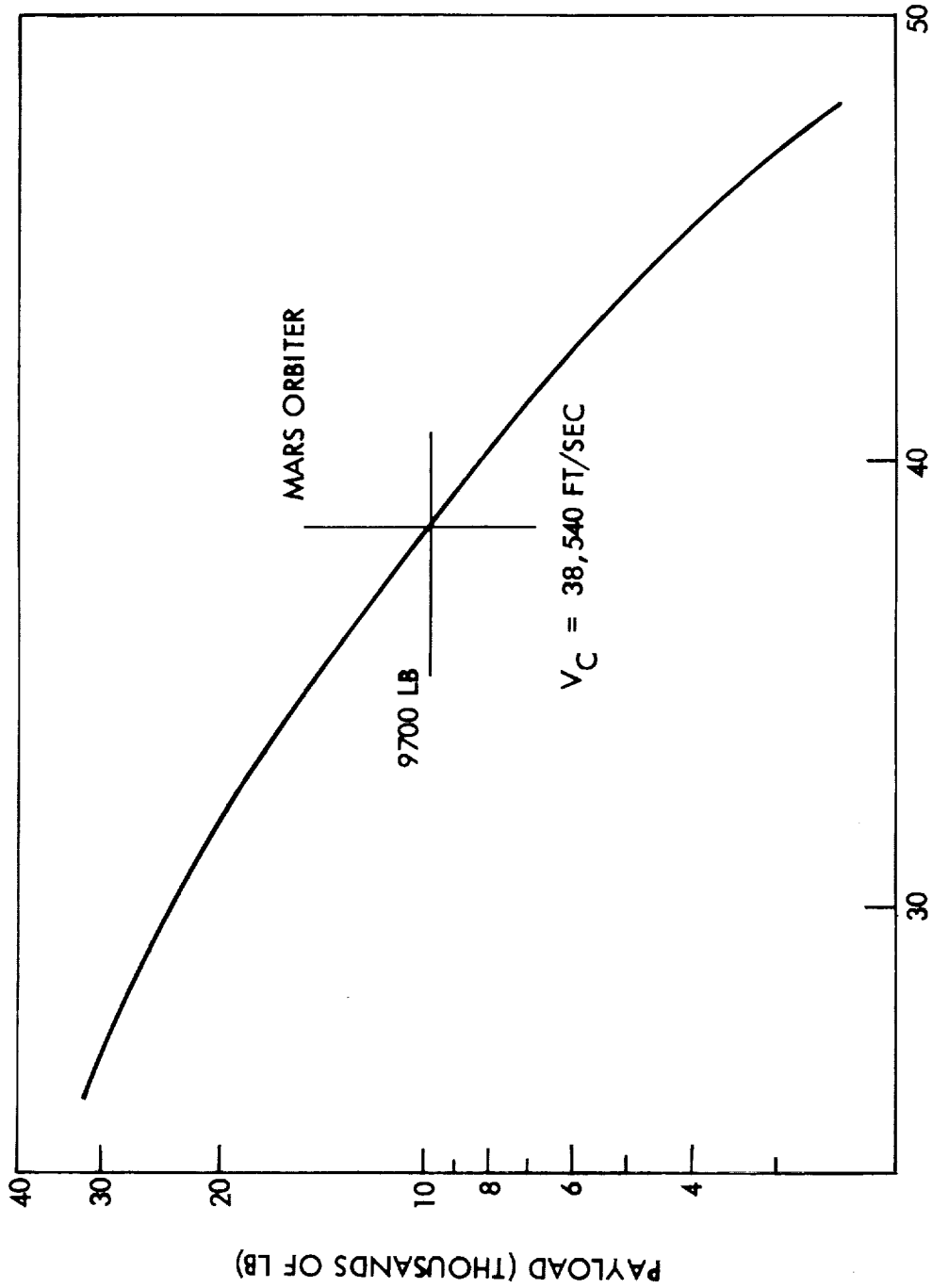


Fig. 17 Titan IIID/Centaur Performance Capability

The first parameter examined in the analysis was the engine thrust/weight (T/W) effect for each propellant and vehicle combination. The result was a thrust value of 5,000 pounds selected for the pump-fed systems and 3500 pounds for the pressure-fed systems, with throttling to 500 pounds assumed as a requirement.

Engine data were solicited from the supporting engine manufacturers for the chosen engine sizes and types and the resulting inputs used in arriving at assumed values for mixture ratio and I_{sp}. These values, together with the favored vehicle cruise orientation, are shown in Table 27.

Table 27

VEHICLE ASSUMPTIONS

Propellant	Feed Type	Thrust (lb)	Mixture Ratio (O/F)	Expansion Ratio (ϵ)	I_{sp} (sec)	Orientation
F_2/H_2	Pump	5,000	12	100	464	Sun on capsule
$FLOX/CH_4$	Pump	5,000	5.25	100	402	Sun on capsule
OF_2/B_2H_6	Press.	3,500	3	60	397	Sun on capsule
N_2O_4/A-50	Press.	3,500	1.6	60	311	Sun on tanks

Sizing evaluations were performed in order to determine initial propellant loadings for the detailed analyses. A typical example of the sizing results is shown in Fig. 18 for F_2/H_2 propellants. From this, and similar results for the alternate propellants, the initial propellant loads chosen were:

Propellant	Propellant Load (lb)
F_2/H_2	3600
$FLOX/CH_4$	4000
OF_2/B_2H_6	4100
N_2O_4/A-50	4900

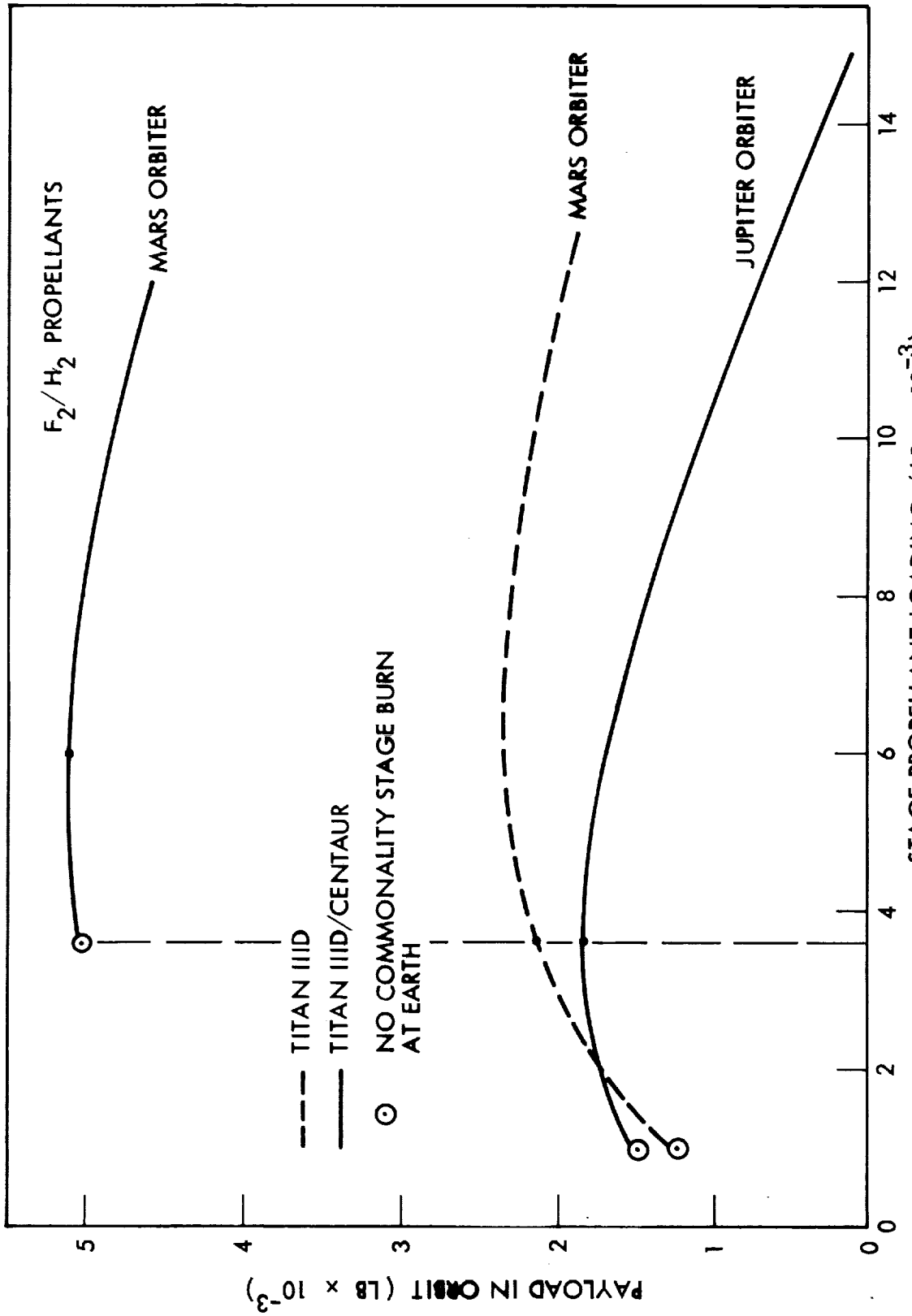


Fig. 18 Commonality Stage Payload Vs Propellant Loading

In addition it was decided that propellant loads of 6,000 lb for F_2/H_2 and FLOX/CH_4 would be evaluated for the lunar lander mission and for some ascent lunar missions.

4.5.3 Mission and Stage Requirements

Alternate missions and associated propulsion stage requirements are summarized in Table 28. These missions represent two distinct categories of use for a propulsion stage – one where there is long term propellant storage such as for planetary orbiters and landers, and the other where the stage is expended during departure from earth such as for flyby missions.

Table 28

CHARACTERISTICS OF ALTERNATE MISSIONS

Mission	Launch Year	Injection Velocity Departing Earth (ft/sec)	Time to Final Burn (days)	Stage ΔV After Injection at Earth (ft/sec)	Number of Burns Expected
Mars Orbiter	1973	38,540	205	6,950	4 or 5
Lunar Cargo	1975	36,027	3	9,006	4
Venus Orbiter	1976	38,339	170	7,200	3
Jupiter Orbiter	1980	47,361	900	6,726	4
Jupiter Flyby	1975	46,600	hours	0	1
Solar Probe (0.2 AU)	1977	55,000	hours	0	1
*Grand Tour of Jovian Planets	1977	47,900	hours	0	1
*Earth Synchronous Equatorial Orbiter	1975	33,660	hours	5,991	1

*These missions were added to the matrix specified, but no detailed description or analyses are provided.

4.5.4 Stage Design and Analysis

A detailed design analysis of the commonality stage was conducted with F_2/H_2, FLOX/CH_4, OF_2/B_2H_6, and N_2O_4/A-50 propellants using the Mars orbiter mission as the basis for a design optimization. Preliminary configuration layouts were refined and structural and weight analyses conducted. Engine data supplied for this task by the supporting rocket engine companies was evaluated and engine performance, weight, and design conditions were selected for the analysis. A detailed thermodynamic analysis was made in which all configurations were modeled, energy balances determined, and a system optimization conducted for each mission and propellant combination. The performance of each propulsion module for each mission and propellant combination was then determined and final designs including weights of each of the elements of the propulsion module, were established.

A summary of optimized insulation thicknesses and maximum tank pressures is presented in Table 29 for all combinations of propellant and missions. In all cases the propellant tanks are flown non-vented throughout the mission.

An example of a typical commonality stage design is presented in Fig. 19 for FLOX/CH_4 propellants. A summary of weights for this stage when used as a Mars Orbiter is shown in Table 30. The table also summarizes weights for the remaining propellants.

4.5.5 Performance Results

Payloads were computed for each combination of mission and propellant, and the results are summarized in Table 31. Additional missions, not analyzed, will also be attractive with the Titan IIID/Centaur and commonality stage combination. These include flybys of Saturn, of Mercury by way of Venus, of comets and of asteroids. Figure 20 shows the direct injection capability of the commonality stage when flown on Titan IIID with or without Centaur.

Table 29

SUMMARY OF INSULATION THICKNESS AND MAXIMUM TANK PRESSURES

Mission	F_2/H_2				FLOX/Methane				OF_2/B_2H_6				N_2O_4/A-50			
	F_2		H_2		FLOX		CH_4		OF_2		B_2H_6		N_2O_4		A-50	
	Thk*	Press.†	Thk	Press.	Thk	Press.	Thk	Press.	Thk	Press.	Thk	Press.	Thk	Press.	Thk	Press.
Mars Orbiter (Baseline)	1/2	44	2 1/4	91	1/2	65	1/2	31	1/2	241	1/2	155	1/2	160	5/8	155
Venus Orbiter	1/2	213	3	94	1/2	210	1/2	48	1/2	228	1/2	155	1/2	174	1/2	155
Lunar Cargo	1/2	33	1	25	1/2	33	1/2	33	1/2	166	1/2	166	1/2	158	1/2	167
Jupiter Orbiter	1/2	33	1	166**	1/2	41	1/2	32	5/8	226	1/2	165	5	219	4	190
Mars Orbiter w/ Titan IIID	5/8	44	2 3/8	127**	5/8	51	1/2	35	1/2	216	1/2	166	1/2	166	1/2	166
Mars Orbiter W_p = 6000 lb	1/2	44	2 7/8	98	1/2	43	1/2	35								

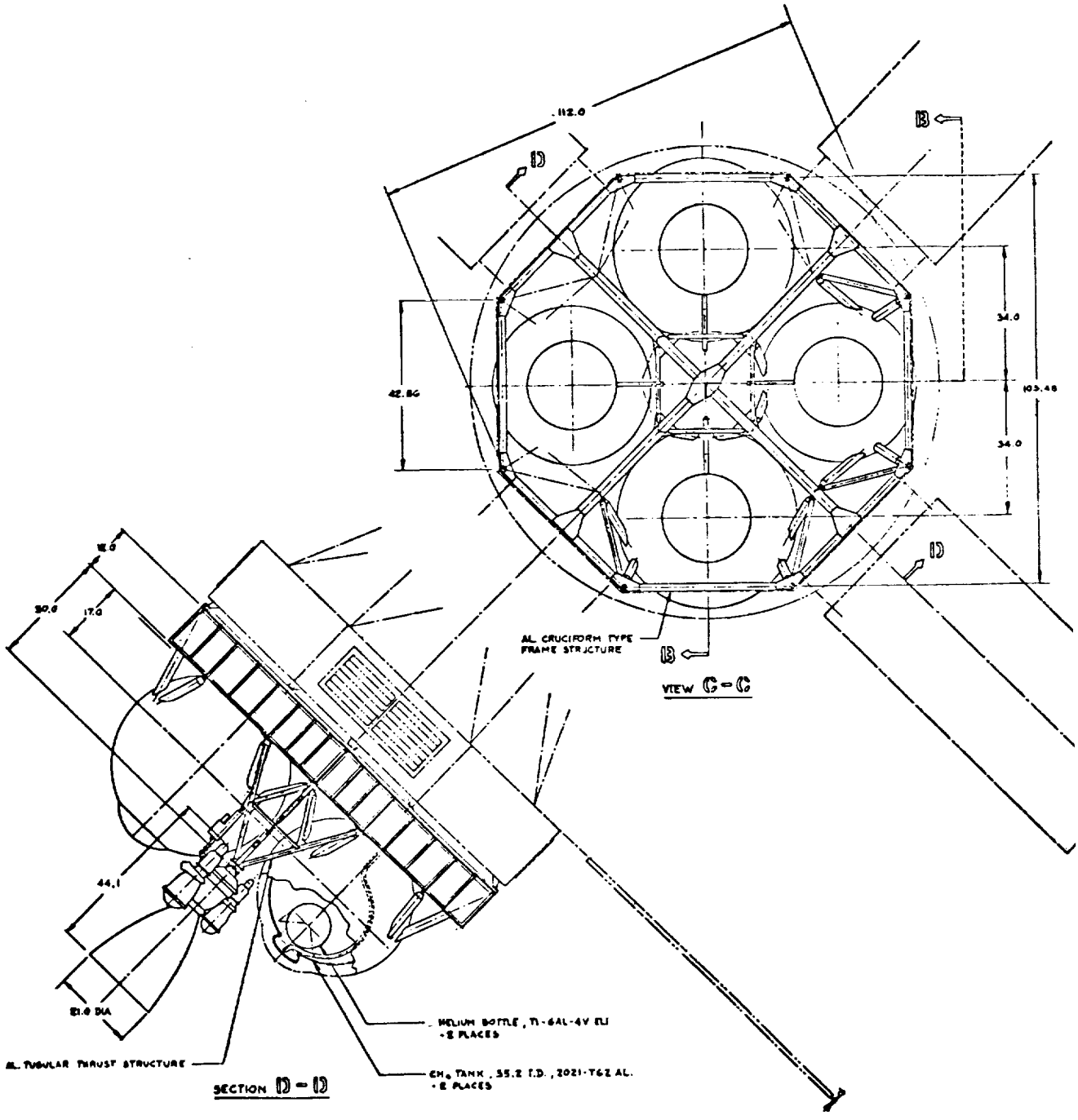
*Thk = Insulation thickness, inches
**Above design pressure.
+Press. = max. Tank Pressure, psia

Table 30

WEIGHTS FOR BASELINE COMMONALITY (MARS ORBITER) STAGE

(Weights in Pounds)

Item	F_2/H_2	FLOX/CH_4	OF_2/B_2H_6	N_2O_4/A-50
Structure	251	177	178	210
Propellant Feed Assembly	419	381	406	381
Pressurization System	27	27	59	79
Engine System	98	98	153	151
Contingency	79	68	80	82
Residuals	121	86	100	93
Performance Reserve	28	31	31	34
Inert	1023	868	1007	1030
Impulse Propellant	3622	4044	4085	4868
Propulsion Module	4645	4912	5092	5898
Payload	5055	4788	4608	3802



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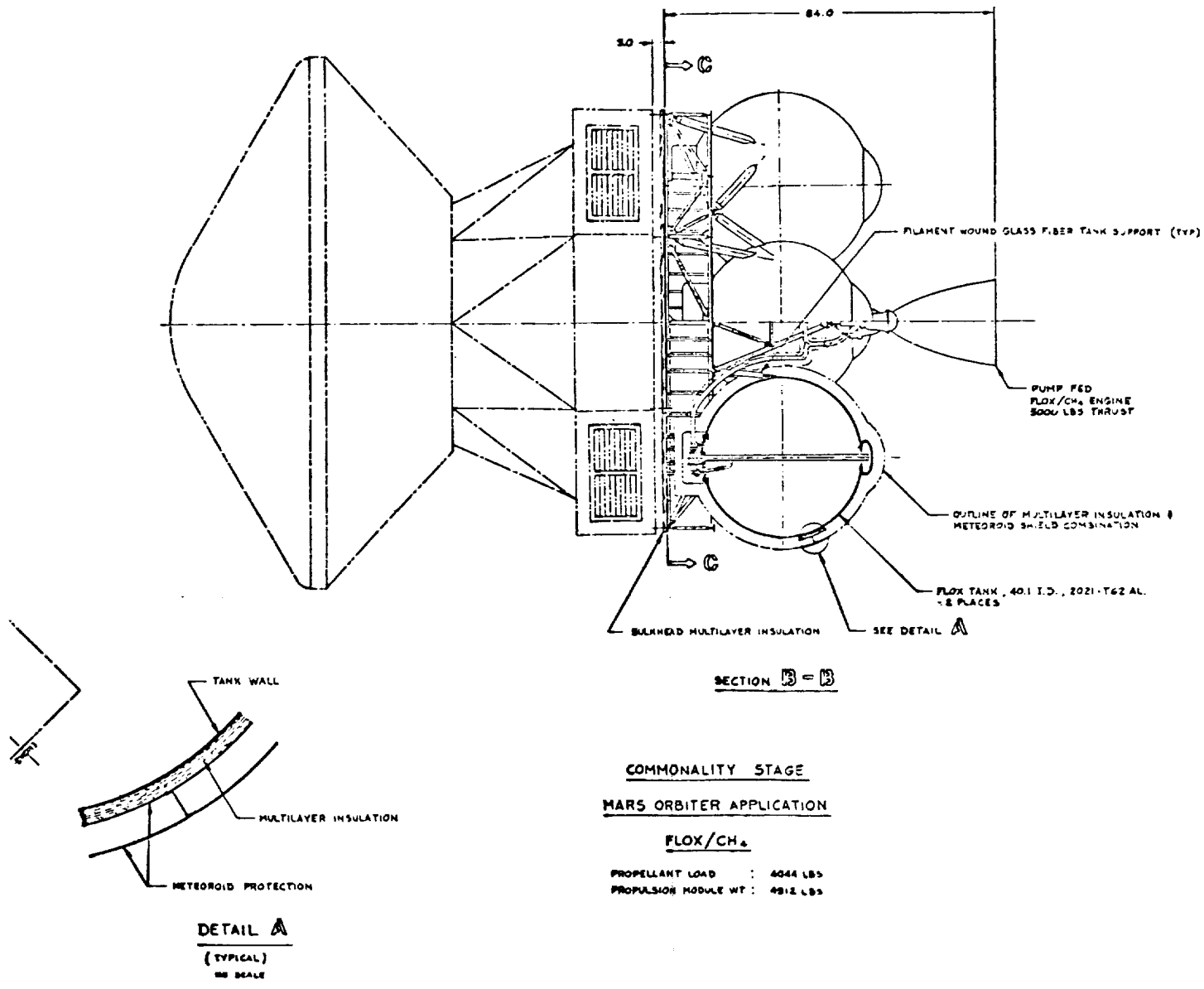

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Fig. 19 FLOX/CH_4 Commonality Stage Design

Table 31

COMMONALITY STAGE PAYLOAD CAPABILITIES SUMMARY

Mission	Payloads (Pounds)			
	F_2/H_2	$FLOX/CH_4$	OF_2/B_2H_6	N_2O_4/A-50
Mars Orbiter	5055	4788	4608	3802
Venus Orbiter	4779	4505	4352	3555
Lunar Cargo	3434	3170	2995	2317
Jupiter Orbiter	1769	1661	1479	784
Mars Orbiter W/Titan IIID	2255	2120	1945	1328
Mars Orbiter – W_p = 6000 lb	5370	5040	–	–
Solar Probe	1830	1590	1460	890
Jupiter Flyby	4700	4360	4140	3420
Grand Tour of Jovian Planets	4090	3770	3570	2880

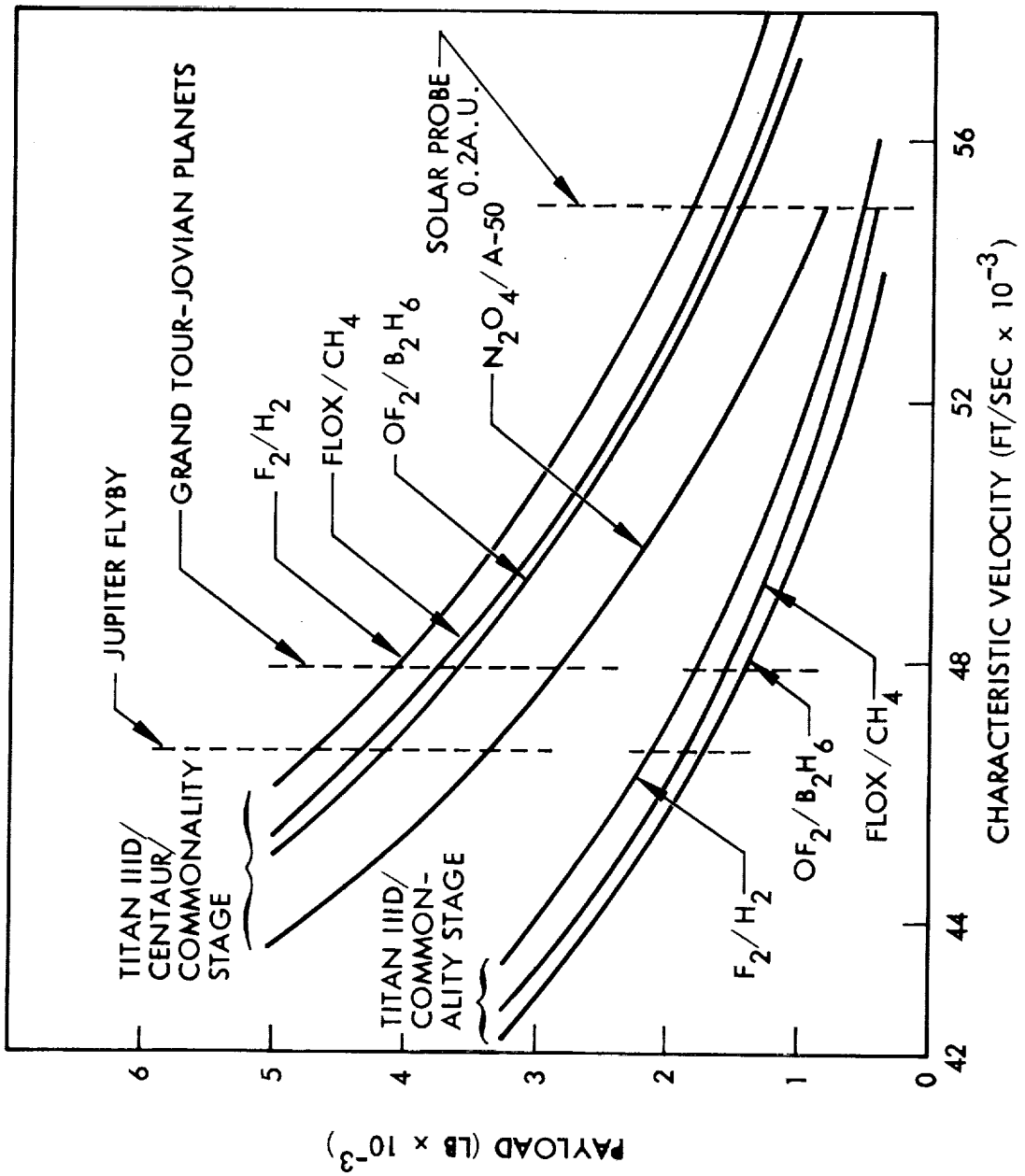

Fig. 20 Direct Ascent Payload Capability

4.5.6 Commonality Stage Conclusions

The commonality concept has been verified in the design analysis showing that a broad range of missions can be accomplished with the same basic system. F_2/H_2 shows the greatest sensitivity to this concept but can still be readily accommodated. Table 32 shows the mission peculiar optimized weight and design parameters based on a vehicle fully optimized for the Mars Orbiter mission. The small variations in the parameters are indicative of the commonality concept. With the exception of the hydrogen tank pressure for the Jupiter Orbiter and for the Mars Orbiter with Titan IIID, all hardware systems are common. The tank pressure perturbation increases the tank weight 45 pounds based on the Jupiter mission and 20 pounds based on the Mars Orbiter/Titan IIID mission. Only meteoroid shields and insulation thickness require a mission peculiar design. Consequently:

- A commonality stage can quite effectively perform a wide range of space missions
- Both cryogens and space storables are attractive in a commonality stage:
  - F_2/H_2 gives best performance
  - FLOX/CH_4 and OF_2/B_2H_6 are less sensitive
- N_2O_4/A-50 stage payloads are relatively poor and active propellant heating may be required for outer planet missions

Table 32

F_2/H_2 COMMONALITY STAGE CHARACTERISTICS

Mission	Payload (lb)	Impulse Propellant (lb)	Meteoroid Shield (lb)	Pressurization System (lb)	Tank Operating Pressure (psi)		Insulation Thickness (in.)		Initial Ullage (%)	
					F_2	H_2	F_2	H_2	F_2	H_2
Mars Orbiter	5055	3622	77	27	44	91	1/2	2-1/4	2	15
Venus Orbiter	4779	3620	72	27	213	94	1/2	3	2.4	22.4
Lunar Cargo	3434	3632	33	26	33	25	1/2	1	2	23.3
Jupiter Orbiter	1769	3635	130	27	33	166*	1/2	1	2	10.4
Mars Orbiter w/Titan IIID	2255	3622	77	26	44	127**	5/8	2-3/8	2	9.7

*Exceeds design pressure of commonality stage and imposes a 45 lb increase in tank weight.
**Exceeds design pressure of commonality stage and imposes a 20 lb increase in tank weight.

Section 5
TECHNOLOGY DEVELOPMENT REQUIREMENTS

There is need for continuing the development of fluorine and fluorinated oxidizer technology in all aspects. The primary requirements are related to thermal storage and prevention of leakage. These requirements fall in the following categories:

(1) Requirements and problems associated with handling and storage of propellants throughout all operational phases. Development needs are identified for:
- Simple, efficient, condensation-free ground thermal conditioning systems
- Simple, lightweight, economical surface coatings, with α/ϵ comparable with Lockheed Optical Solar Reflector (OSR), for use on large surfaces
- Evacuated multilayer insulation for use in planetary atmospheres
- Further development of the structural requirements and evaluation of the efficiency of multilayer superinsulation in thicknesses from 2 to 5 in. (this item is required for hydrogen systems only)
- Demonstration of venting times for multilayer insulation under simulated launch conditions

(2) Requirements associated with the continued development of sensors, leakage control, lightweight tanks, and structural components. Work should be continued on:
- Lightweight tanks and tank materials compatibility
- Lightweight, low-heat-leak support struts
- Low-heat-leak tank penetrations
- Materials compatibility
- Leak-free electrical feed-throughs for fluorinated oxidizers
- Leak-free fluorinated oxidizer storage and pressurization systems
- Sensors for fluorinated oxidizer management
- Devices for sensing propellant partial pressure

Technology development requirements related to rocket engine systems were not subjects for detailed analyses but were determined from inputs supplied by the supporting engine companies. These include:

- Low-flow, high ΔV fuel pumps and small turbines
- System tests to confirm realistically attainable performance
- Optimum engine cooling technique development for each propellant
- Throttling, restart, and control of small engines

Section 6
CONCLUSIONS

The following conclusions were reached during the conduct of Phases I and II of the Space Storable Propellant Selection Study:

(1) Space-storable and deep cryogenic propellants have attractive performance potential for a wide range of space missions requiring thrust levels between about 1,000 and 50,000 lbf. No serious propellant storage problems are foreseen.

- Space-storables outperform earth-storables by from 21 to 112 percent, and F_2/H_2 outperforms earth-storables by from 33 to 126 percent for the missions considered as attractive candidates.
- Fluorine/Hydrogen provides the best performance for all mission applications studied
- Space-storable propellants provide high performance and are thermally less sensitive than hydrogen
- Specific impulse is the predominant factor in propellant/system performance
- A space-storable or F_2/H_2 engine system in the 5,000- to 10,000-lb-thrust range, used singly or clustered, could meet the performance requirements of most of the missions analyzed, while incurring a stage weight penalty never exceeding three percent of the nominal.

(2) Space-storables and deep cryogens are relatively insensitive to variations in spacecraft orientation, mission length, and degradation of insulation and surface coatings.

- All propellants studied can be flown nonvented for a wide range of probe, orbiter, and lander missions.
- A preferred spacecraft orientation is desirable but not mandatory. Protecting the nonvented propulsion system of the Mars Orbiter from the sun by orientation or local shadow shielding resulted in a reduction of insulation thicknesses to 2 in. or less, while increasing payload capability.

- Hydrogen is more sensitive to the space thermal environment than space-storable propellants because external insulation surface temperatures on H_2 tanks cannot practically be reduced to the H_2 liquid temperature. Even with controlled orientation and shadown shielding the temperature across the insulation will always be at least 100 degrees F. For space-storable propellants a temperature difference across the insulation of less than 25 degrees F is easily achieved and in most cases a differential of zero degrees F can be maintained.

(3) Space-storable propellant feed systems simplicity is similar to earth-storables.

(4) The concept of a commonality stage to be flown on a wide range of missions is feasible and very attractive for either F_2/H_2 or space-storable propellants.

(5) Operational Considerations include:

- All propellants studied can be loaded off-pad and maintained non-vented by active thermal control until launch.
- A helium purge of the insulation is required during ground hold for the H_2 propellant and dry nitrogen or dry air for the space storables.
- A typical flight-type multilayer insulation installation will vent adequately in a typical boost trajectory following proper ground-hold conditioning. Ascent heating of ullage gas will not cause unacceptable pressure rises.

Section 7
RECOMMENDATIONS

The following recommendations are made following the completion of Phases I and II of the Propellant Selection study:

(1) Continue the development of fluorine and fluorinated oxidizer technology in all aspects, as outlined in Section 5

(2) Perform a detailed analysis of operation of space-storable and cryogenic systems in low orbits about warm planets

(3) Continue the development of better propellant-property data, particularly vapor enthalpy, internal energy, and heat-of-vaporization as a function of pressure

(4) Perform a critical analysis of all space-storable (and related) study and technology work accomplished to date in order to narrow the matrix of space-storable propellant combinations, blends, etc., and to establish reliability and feasibility of pump vs pressure feed, competitive cooling methods, practical chamber pressure design points, and throttleability characteristics.

(5) Determine the potential for a space-storable or F_2/H_2 commonality stage in the 1975 to 1985 time period, recognizing the need for an early decision and start on a new engine system.

Section 8
REFERENCES

1. Final Report, Propellant Selection for Spacecraft Propulsion Systems, Lockheed Report No. K-19-68-6, Volume I – Results, Conclusions, and Recommendations (NASA CR 96743), Volume II-Missions and Vehicles (NASA CR 96988), Volume III Thermodynamics and Propulsion (NASA CR 96740) dated 30 August 1968

2. Final Report – Phase II, Propellant Selection for Unmanned Spacecraft Propulsion Systems, Lockheed Report No. K-21-69-9, Volume I – Results, Conclusions, and Recommendations (NASA CR 105201), Volume II – Analysis of Propellant Sensitivity, Secondary Propulsion, and Ground Operations (NASA CR 105202), Volume III – Study of Propulsion Stage Commonality and Attitude Control Systems Requirements (NASA CR 105203)